PROCEEDINGS -

RENDEZVOUS AND PROXIMITY OPERATIONS WORKSHOP

HOSTED BY

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VOLUME III - PRESENTATIONS FROM SESSIONS 5B THROUGH 8

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27 MARCH 1985

27 March 1985

VOLUME III - PRESENTATIONS FROM SESSIONS 5B THROUGH 8

Due to the large quantity of information contained in the eleven sessions of the Rendezvous and Proximity Operations Workshop, the presentation material has been assembled into three volumes for the Proceedings. These three volumes (Volumes II - IV) are in addition to the Executive Summary (Volume I), which was published and distributed on 27 February 1985.

- Volume I - EXECUTIVE SUMMARY, 27 February 1985
- Volume II - PRESENTATIONS FROM SESSIONS 1 THROUGH 5A
- VOLUME III - PRESENTATIONS FROM SESSIONS 5B THROUGH 8
- VOLUME IV - PRESENTATIONS FROM SESSION 9 THROUGH 11

An itemized list of the presentations and authors precedes each of the Sessions in this volume.

SESSION 5B - MAN AND/OR MACHINE OPERATIONS

5B-1. "AUTOMATED RENDEZVOUS AND DOCKING SYSTEM" - JOHN TIETZ/ MARTIN MARIETTA AEROSPACE

5B-2. "AUTOMATED SATELLITE SERVICING ON ORBIT" - HANS MEISSINGER AND E. MEDLER/TRW

5B-3. "A JAPANESE EFFORT IN SPACE ROBOTICS" - K. MACHIDA, M. ONO, AND Y. TODA/ELECTROTECHNICAL LABORATORY

5B-4. "TELEPRESENCE SYSTEMS: ANALYSIS AND NEUTRAL BUOYANCY VERIFICATION" - DAVID AKIN/MIT SPACE SYSTEMS LABORATORY

5B-5. "INCREASED INTELLIGENCE AND AUTONOMY IN SPACE OPERATIONS" - A. BEJCZY, C. RUOFF, AND S. SZIRMAY/JPL

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AUTOMATED RENDEZVOUS AND DOCKING SYSTEM

PRESENTATION AT THE
OMV PROXIMITY OPERATIONS CONFERENCE
JOHNSON SPACE CENTER
HOUSTON, TX

BY
JOHN C. TIETZ
MARTIN MARIETTA DENVER AEROSPACE

OCTOBER 17, 1984

ACKNOWLEDGMENTS

THE WORK DESCRIBED HERE WAS PERFORMED UNDER NASA CONTRACT NAS8-34679. SIGNIFICANT BENEFITS WERE DRIVED FROM PREVIOUS RELATED WORK PERFORMED UNDER MARTIN MARIETTA DENVER AEROSPACE IR&D TASK D-11R.

WHAT WE HAVE DONE

- PRODUCED CONCEPTUAL DESIGNS FOR THREE AUTONOMOUS VIDEO RENDEZVOUS AND DOCKING SYSTEMS
- SIMULATED THESE SYSTEMS WITH A COMPUTER
- CONDUCTED HARDWARE SIMULATIONS OF THE BEST SYSTEM
- PRODUCED A SECOND-GENERATION VERSION OF THIS SYSTEM WITH IMPROVED PERFORMANCE

In the tests we ran, the video guidance system performed reliably with inertially stable targets and with slowly tumbling targets. The three-point target docking aid concept could be implemented without flash lamps if necessary. For example, painted spots with unique spectral signatures might be used with a color camera. The basic features of the control loop might also be used with a different method of sensing relative position and attitude.

The video system did not perform well with rapidly tumbling targets, primarily because the docking aid periodically rolled out of sight behind the body of the target spacecraft. This problem could be cured by adding multiple docking aids.

The mathematical model used in the Kalman filter in this system can be adjusted with data from any source, so it is not necessary to switch guidance systems when the video system becomes active; a single system can be used for all phases.

CONCLUSIONS

- o A VIDEO GUIDANCE SYSTEM OF THIS TYPE APPEARS PRACTICAL
- o FEATURES OF THIS SYSTEM COULD BE USED IN OTHER SYSTEMS
- o TUMBLING TARGETS REQUIRE MULTIPLE DOCKING AIDS FOR TUMBLE RATES OVER ONE DEG/S
- o THE KALMAN FILTER CAN ACCEPT AUXILIARY INFORMATION FROM ANY SOURCE, PERMITTING AN INTEGRATED GUIDANCE SYSTEM FOR ALL PHASES

DESIGN REQUIREMENTS

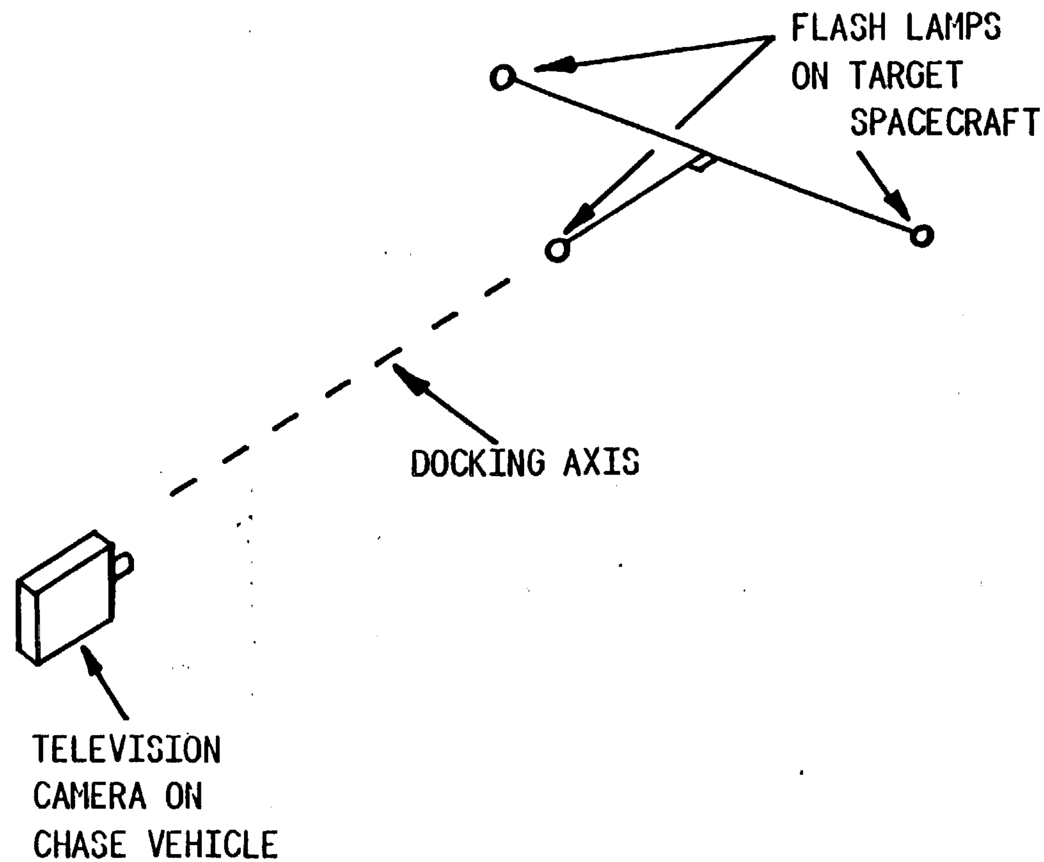
- SYSTEM WORKS WITH COOPERATIVE TARGET
- TARGET CANNOT MANEUVER AND MAY BE TUMBLING
- SYSTEM USES VIDEO GUIDANCE
- SYSTEM HANDLES RANGE OF 300 METERS TO CONTACT

The docking aid comprises three flashing lights. The flash lamps are sequenced so that only one is on at a time. This allows the system to resolve image ambiguity. A standard television camera can be used, but solid-state cameras have advantages.

The image of the docking aid uniquely defines relative position and attitude. A coarse interpretation, accurate at distances where perspective effects can be ignored, can be produced quickly from the image-plane coordinates of the lamps. The required calculations can be done in approximately 50 lines of Fortran with no loops.

The coarse interpretation can be refined by a simple iterative technique. Each iteration removes approximately 90% of the remaining error. After a single iteration errors in measuring the coordinates of the lamps in the image tend to dominate the total error.

DOCKING AID DESIGN



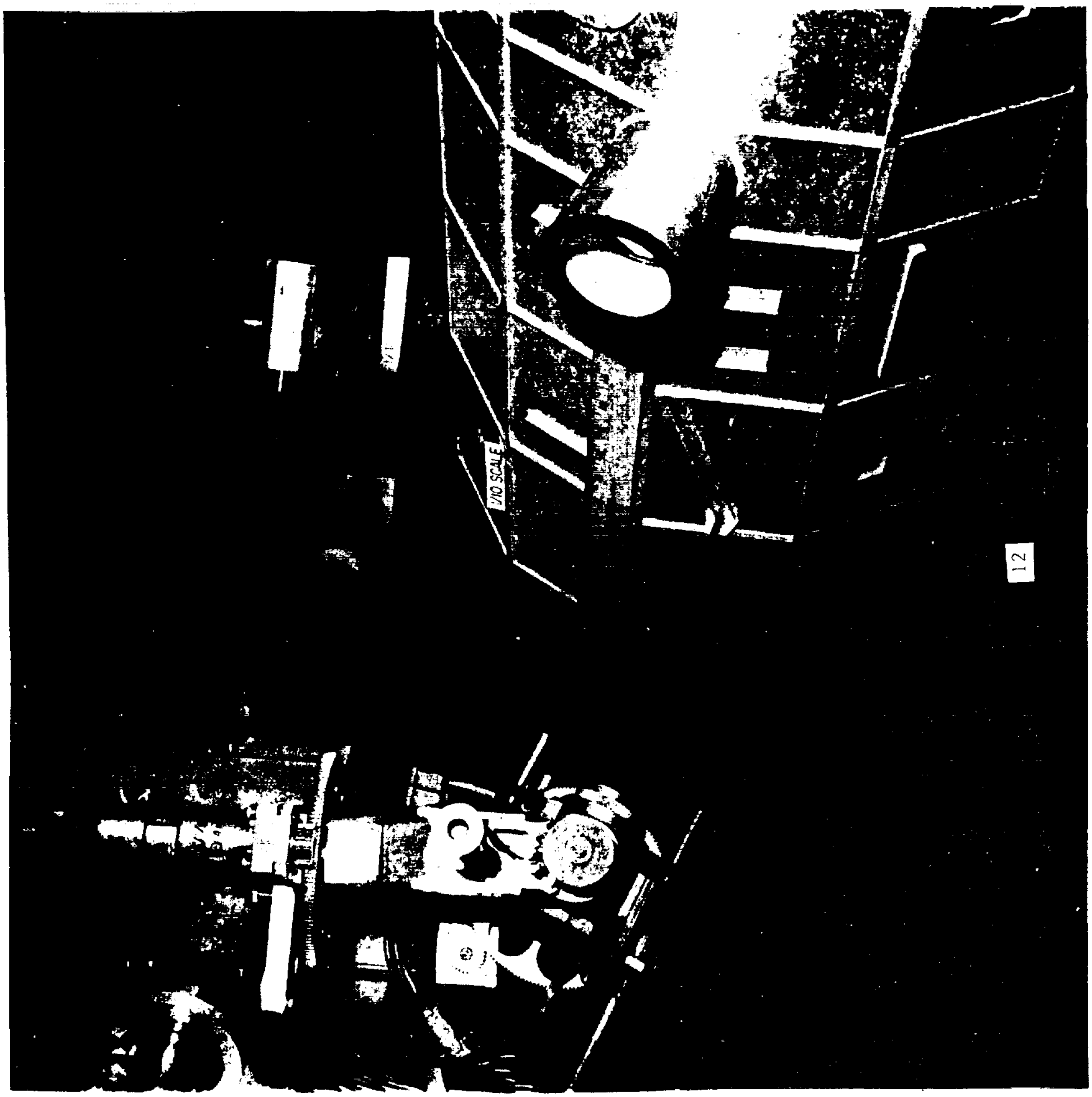
ADVANTAGES OF FLASH LAMPS

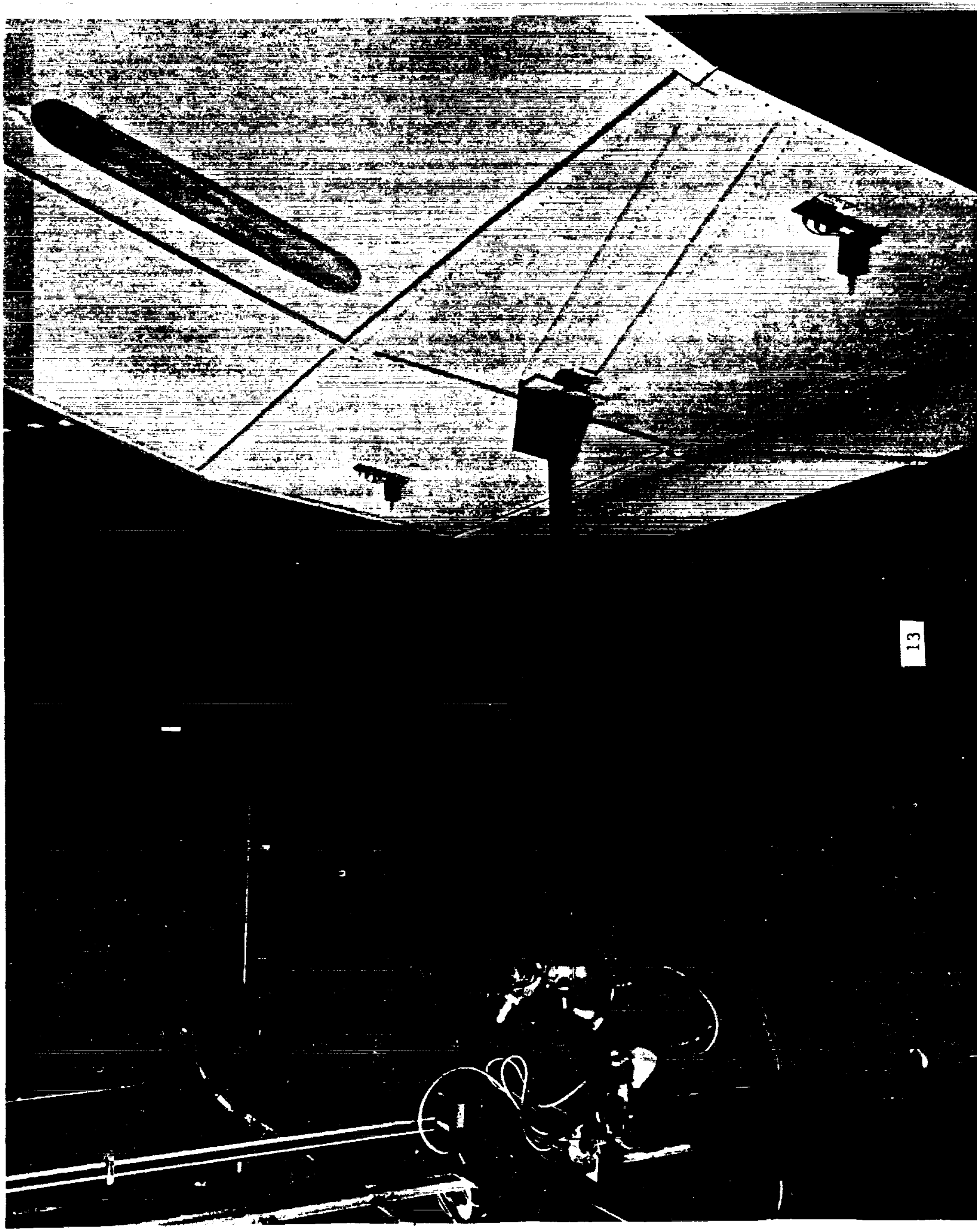
- THE SYSTEM CAN EASILY DETERMINE WHICH LAMP IS WHICH
- HARDWARE CAN COMPUTE LAMP COORDINATES
- SYNCHRONIZED SHUTTER CAN CUT BACKGROUND CLUTTER
- TARGET ILLUMINATION FROM CHASE VEHICLE NOT REQUIRED

CONTROL SYSTEM DESIGN

- TELEVISION IMAGE IS INTERPRETED TO DETERMINE RELATIVE POSITION, ATTITUDE
- THESE MEASUREMENTS ARE COMBINED WITH IMU INFORMATION TO GIVE EQUIVALENT MEASUREMENTS IN A TARGET-CENTERED INERTIAL COORDINATE SYSTEM
- KALMAN FILTER IMPROVES POSITION AND ATTITUDE ESTIMATES, DERIVES VELOCITY AND TARGET TUMBLE PARAMETERS
- CONTROL LOOP OPERATES FROM FILTER'S MATHEMATICAL MODEL, NOT OBSERVATIONS, ALLOWING DEAD RECKONING
- STRATEGY LOGIC GUIDES DOCKING BY ADJUSTING CONTROL LOOP SET POINT. THIS LOGIC OBTAINS ESTIMATES OF POSITION, VELOCITY, ATTITUDES, TUMBLE PARAMETERS, AND PROBABLE ERROR FROM THE FILTER'S MATHEMATICAL MODEL

In our laboratory simulation, scale models of the target spacecraft were used. Each was equipped with a working docking aid. For the smaller models fiber optics were used to pipe light from hidden flash lamps to simulate small lamps. Three models were used to allow simulation of the entire operation in the limited space of the laboratory. The television camera was mounted on a computer-controlled simulator to simulate relative motion. Each coordinate was measured with the camera positioned to provide the same view of the target as the flight system would see. As range decreased, the simulation switched to progressively larger models--first 1/100, then 1/10, then 1/1--to provide increased detail in the image and to minimize the effects of the limited focussing range of the camera lens.


1/10 SCALE
12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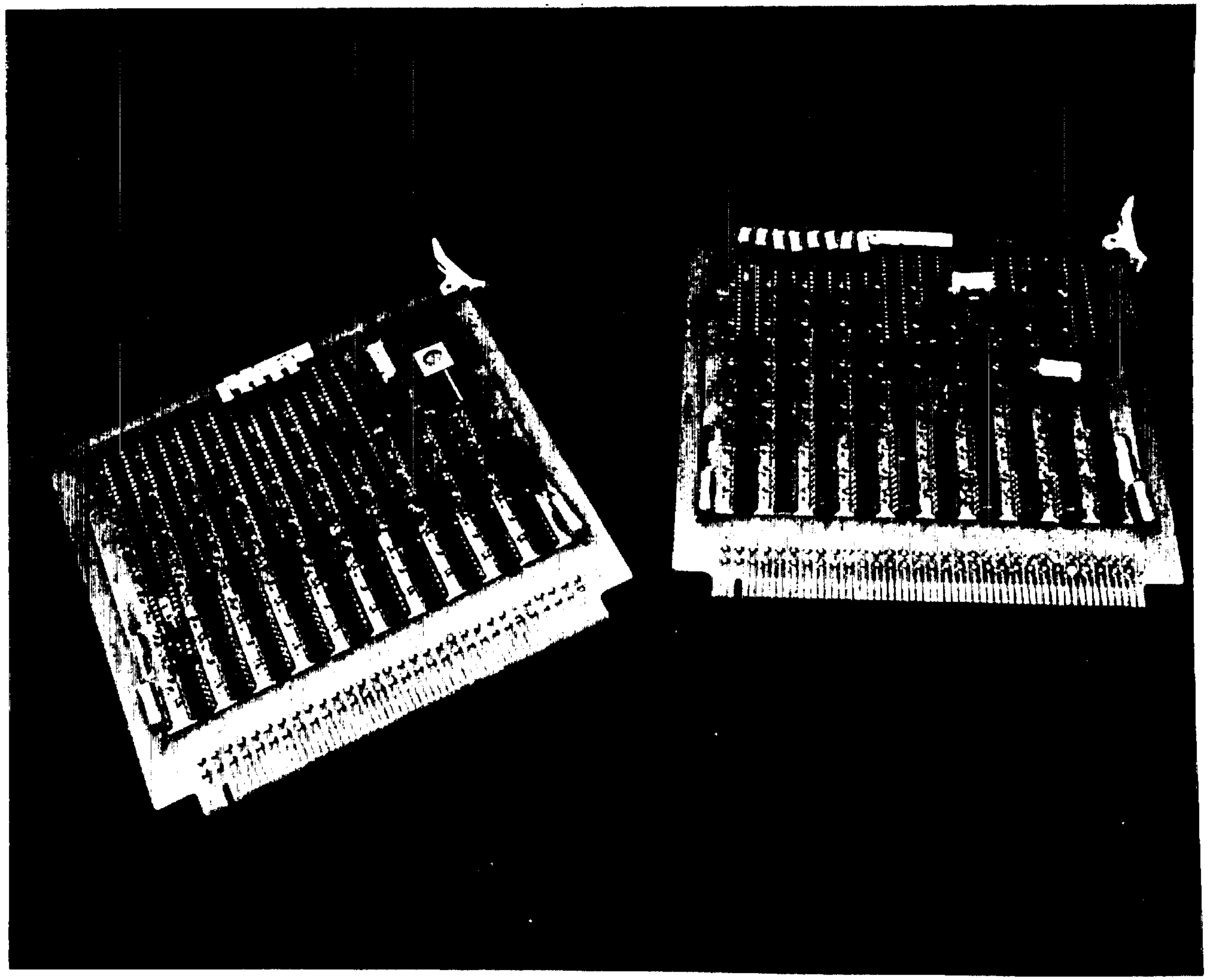
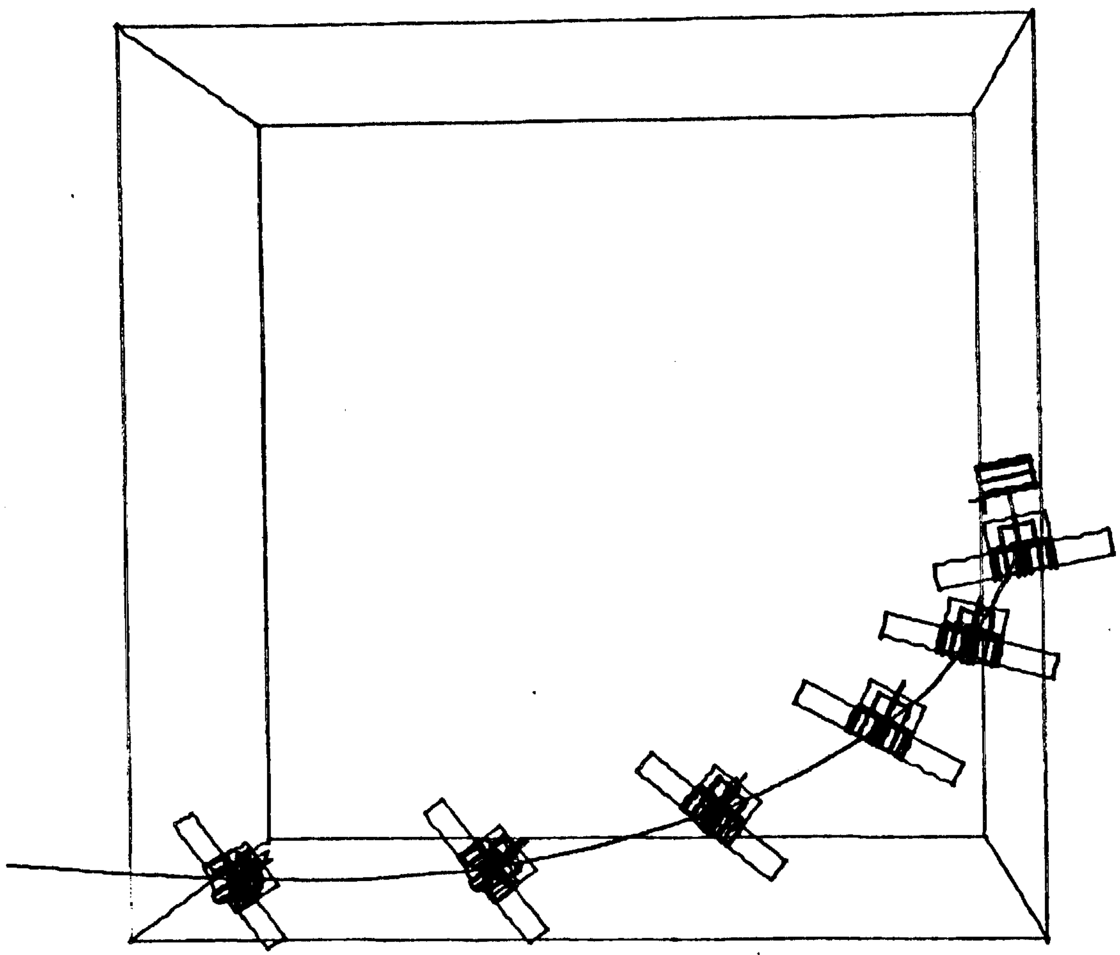
In the simulation the lamp image coordinates were computed with a special-purpose set of electronics. The circuitry calculated the coordinates in real time at the video frame rate. The circuitry then passed the coordinates to the simulation computer along with image-size information, which was used to verify the validity of the measurement.

Video Processing Electronics

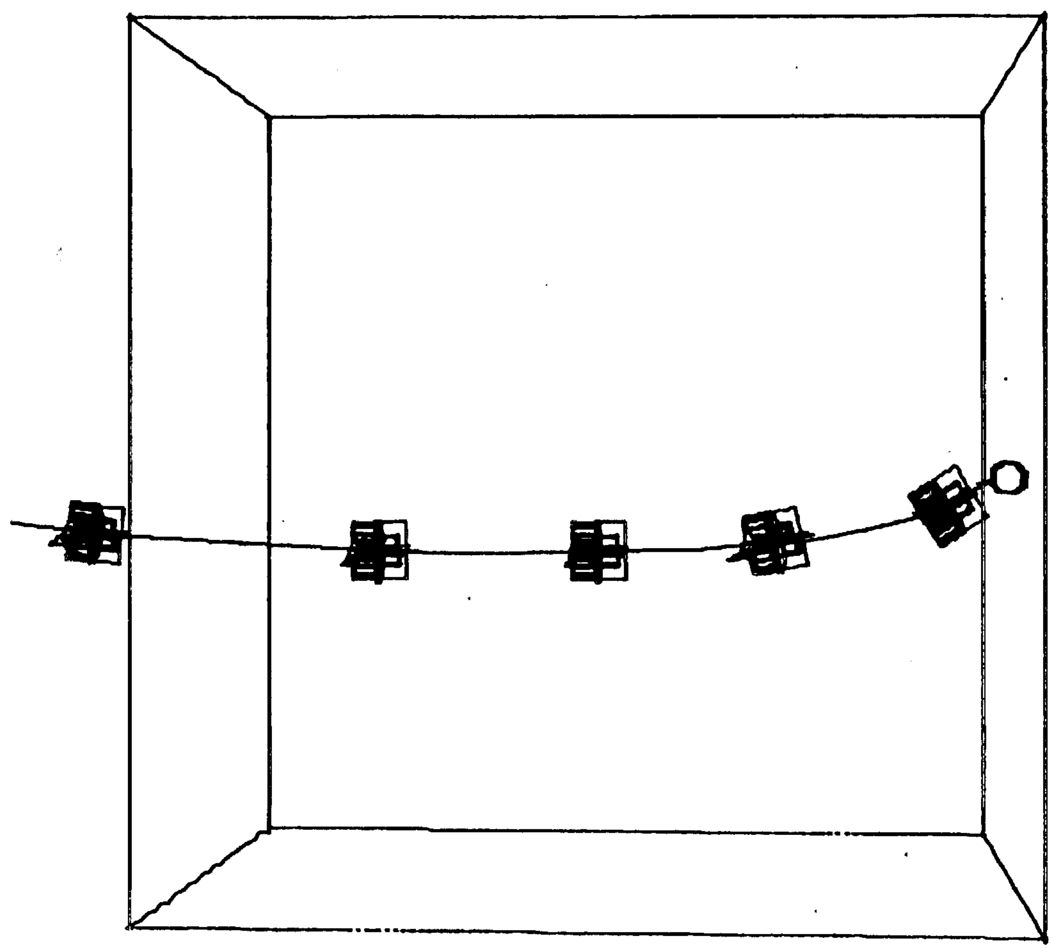


As the simulations progressed, the computer recorded the chase vehicle's position with respect to the target and the attitudes of both spacecraft to allow later analysis of the operation. Simulations were conducted with the target tumbling at various rates about each principal axis. The primary problem at higher rates was docking-aid visibility. Although the Kalman filter permitted dead reckoning for two or three minutes, its position estimates became less accurate with time. The solution to this problem is to provide at least range information when the docking aid is out of sight. This could be done with additional docking aids, an RF system, or with other techniques.

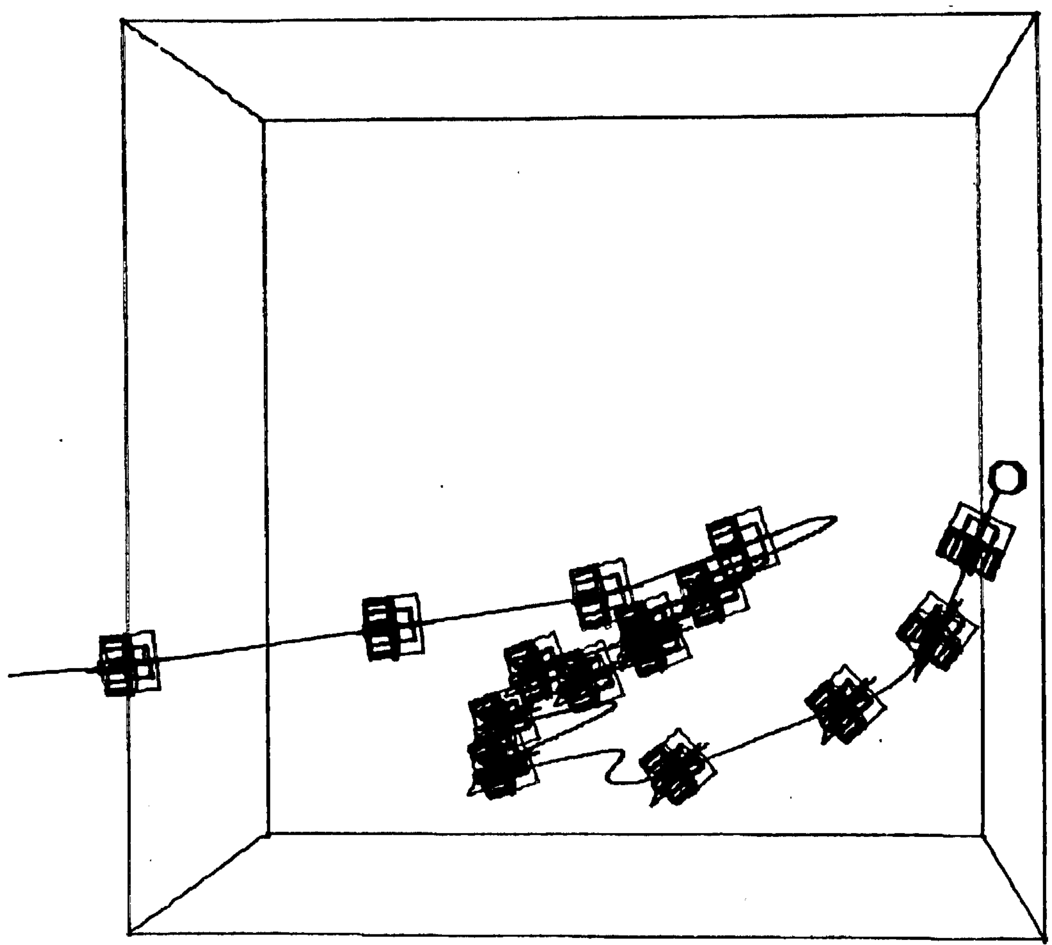
1000-deg/h Yaw Rate



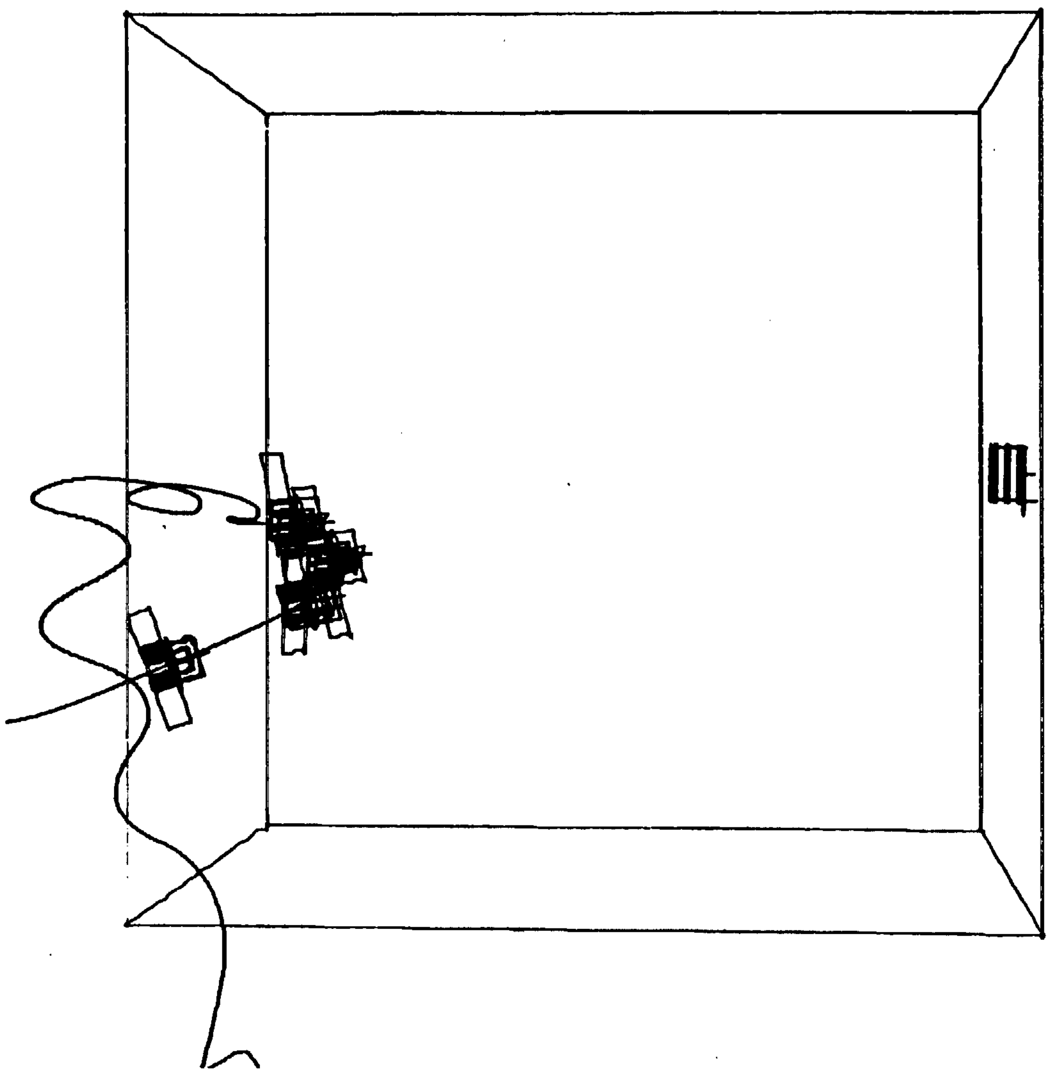
1000-deg/h Pitch Rate



3000-deg/h Pitch Rate



Visibility Problem (3000-deg/h Yaw Rate Shown)



TEST RESULTS

- GOOD PERFORMANCE WITH INERTIALLY STABLE TARGETS
- WORKS WITH TARGET TUMBLE RATES TO ONE DEG/S
- PRIMARY LIMITATION ON WORKING WITH TUMBLING TARGETS IS OBSCURED VIEW OF DOCKING AID
- MULTIPLE DOCKING AIDS COULD SOLVE THIS PROBLEM

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TRW Space & Technology Group

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Z410.1-85-179

AUTOMATED SATELLITE SERVICING ON ORBIT

BY

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19-22 FEBRUARY 1985

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AUTOMATED SATELLITE SERVICING ON-ORBIT

5B-24

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ABSTRACT

A manned Space Station in low earth orbit in the early 1990s makes man's cognitive, observing and manipulative skills available for use in space operations. However, certain satellite servicing functions can be automated such that the best of man's abilities and automated capabilities can be combined for maximum productivity and for minimum crew exposure to hazardous tasks.

Results of a broad-based analysis of current and potential capabilities of telepresence (teleoperation), robotics and artificial intelligence and their use in on-orbit servicing of satellites and Space Station components and attached payloads will be presented; the partitioning of tasks assigned to the crew and to automated systems will be discussed; and attributes of a generic servicing facility for the initial Space Station and projected growth capabilities will be defined. These results are based on a concurrent NASA-sponsored study by TRW of Space Station automation concepts where satellite servicing is one of the major areas of concern.

*Senior System Engineer

AUTOMATED SATELLITE SERVICING

DISCUSSION TOPICS:

- SATELLITE SERVICING REQUIREMENTS
- AUTOMATION REQUIREMENTS
- AUTOMATED TASK CLASSIFICATION
- REFERENCE SERVICING MISSION SCENARIOS
- PRINCIPAL AUTOMATION BENEFITS
- DESIGN IMPLICATIONS OF AUTOMATED SERVICING
- CONCLUSIONS

MAN-MACHINE PARTITIONING CONSIDERATIONS

One of the major objectives of the study was to determine effective combinations of the strongest capabilities of automated functions and of man's functions in performing satellite servicing tasks. The chart lists principal criteria of the strength of machine operations versus human operations. The automated system is capable of performing repetitive operations under predictable conditions and is utilized most effectively where it enhances true productivity and safety (e.g., by tasks which would otherwise require EVA). Man's unique cognitive sensing and manipulative skills and his ability to react to unforeseen situations were the criteria for assigning certain tasks to the crew rather than the automated system. Related experience on Shuttle missions in 1984 highlights this fact: 1) the retrieval and repair by astronauts of the Solar Max Mission (SMM) spacecraft in April 1984 and the recovery of two communications satellites, Palapa and Westar in November 1984.

Related questions addressed by the study include the following: What type of automation or robotics is needed and how will it be used? How much does automation facilitate crew tasks and enhance productivity? How much time saving is achieved? What is the impact on operational safety and what satellite design, standardization and operational requirements are imposed by automated servicing?

Subsequent discussion in this briefing will take issue with these important questions.

MAN-MACHINE PARTITIONING CONSIDERATIONS

MACHINE CRITERIA	MAN CRITERIA
• TIME CRITICAL	• TIME EFFECTIVENESS
• REPETITIVE/PREDICTABLE	• UNPREDICTABLE
• PRECISION	• MOTOR SKILLS
• PRODUCTIVITY ENHANCEMENT	• COGNITIVE ABILITY
• SAFETY ENHANCEMENT	• PATTERN RECOGNITION
• REMOTE LOCATION	• SEQUENCING COMPLEXITY

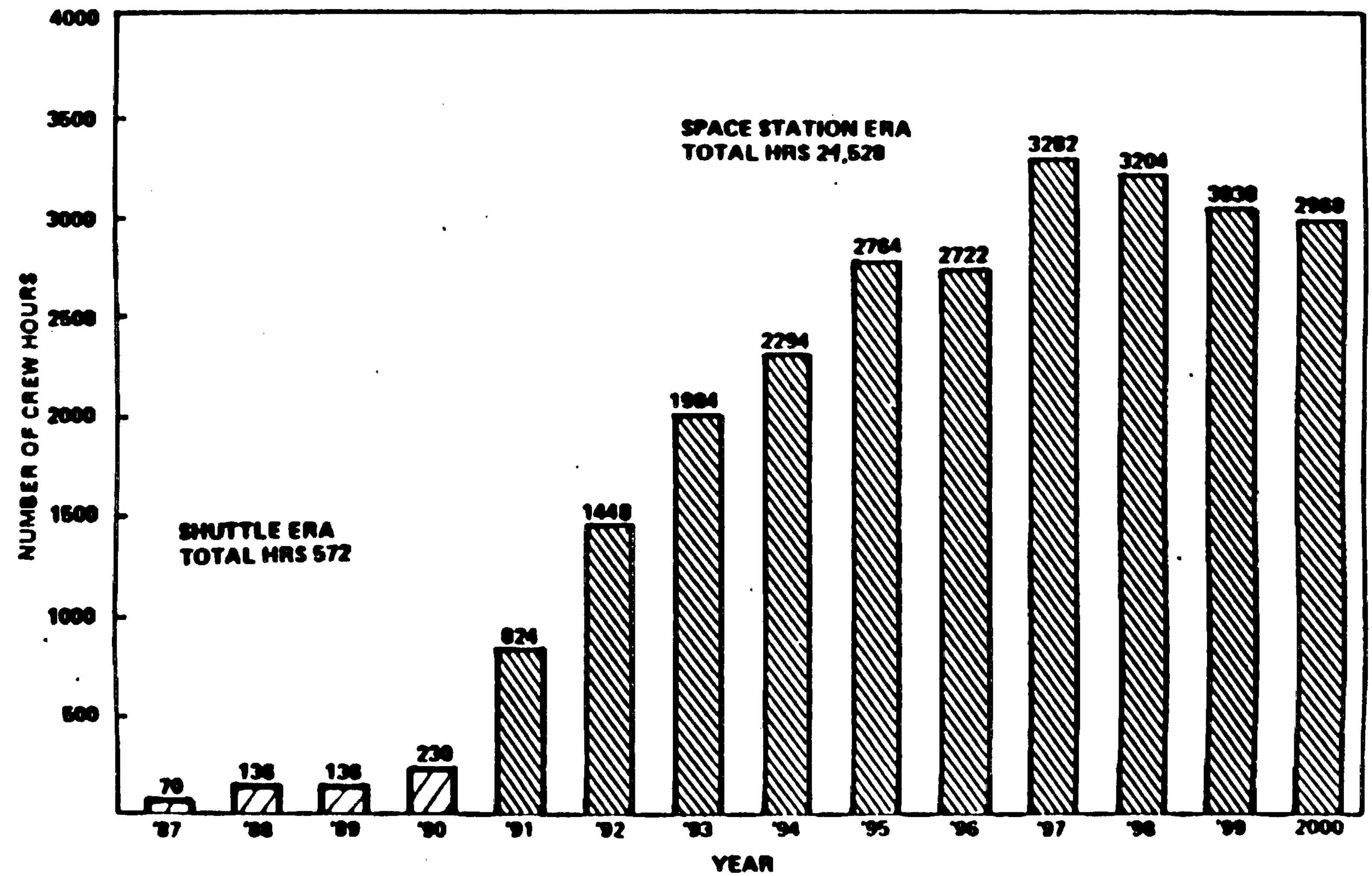
CREW SATELLITE SERVICING ACTIVITY (WITHOUT BENEFIT OF AUTOMATION)

From an analysis of on-orbit satellite servicing requirements in the current NASA mission model a projection of annual crew servicing activities was obtained. As shown in this chart, the crew hours required for servicing increase rapidly during the 1990s after the advent of the IOC Space Station (assumed here in 1991) averaging above 2500 hours annually. Of these about 60 percent are spent on IVA and 40 percent on EVA tasks.

Actually a further sustained increase of crew activities in the late '90s is to be expected, although it is not indicated by data from this analysis of servicing events, which is due, probably, to an increasing uncertainty of mission forecasts.

Analysis of crew servicing activities required in four representative mission scenarios indicates that time savings of about 50 percent are attainable by automated operations (teleoperation and robotics). A significant share of these time savings is attributable to a reduction in EVA activities and the required preparation procedures.

CREW SATELLITE SERVICING ACTIVITY WITHOUT BENEFIT OF AUTOMATION (BY YEAR)



SHUTTLE-BASED AND SPACE STATION, NOMINAL FIGURES

SATELLITE/MISSION CLASSES AND SERVICING FUNCTIONS

The chart shows principal servicing functions to be performed in representative satellite or mission classes. The first three lines list servicing functions on the Space Station (SS) itself, its subsystems or attached payloads. Some of the functions such as inspection, test, checkout, module replacement and resupply of consumables are required in many of the missions. Control of rendezvous and docking or berthing is involved whenever a satellite is being retrieved for servicing or when it is to be serviced remotely by an Orbiting Manouvering Vehicle (OMV) or Orbital Transfer Vehicle (OTV) equipped with a servicer attachment.

Most of the functions shown lend themselves to automated operations, i.e., teleoperated or robotic servicing as an alternative to direct crew involvement. In the case of in-situ servicing, the alternatives are teleoperation by radio command and video feedback or fully robotic operation.

SATELLITE/MISSION CLASSES AND SERVICING FUNCTIONS

SATELLITE OR MISSION CLASS \ SERVICING FUNCTION	DEPLOYMENT	CONTROL RENDEZVOUS & DOCKING	INSPECTION	RETRIEVAL	ORBITAL ASSEMBLY	MAINTENANCE	TEST/CHECKOUT	REPAIR/MODIFIC.	CLEANING, RE-SURFACING	CHANGEOUT/ REPLACEMENT	PREPARE FOR EARTH RETURN	RESUPPLY	EMERGENCY OPS.
SPACE STATION BUILDUP					✓		✓	✓					✓
SS MODULE REPLACEMENT			✓				✓	✓		✓			
SERVICE SS ATTACHED P/L						✓	✓	✓	✓	✓		✓	
SERVICE RETRIEVED SATELLITE	✓	✓	✓	✓		✓	✓	✓	✓	✓	✓	✓	
SERVICE LEO SATELLITE, IN-SITU		✓	✓			✓	✓	✓	✓	✓		✓	✓
SERVICE GEO SATELLITE IN-SITU		✓					✓			✓		✓	
SATELLITE ASSEMBLY ON SS	✓		✓		✓		✓						
SATELLITE LAUNCH FROM SS	✓						✓				✓		
SERVICE SPACE PLATFORM		✓	✓				✓	✓	✓	✓		✓	

REPRESENTATIVE AUTOMATION REQUIREMENTS

This matrix provides an overview of major servicing functions and some of the automated sequences required for their implementation by automatic means. Most of the servicing functions listed will utilize the automated features once they are made available; i.e., there exists a high degree of commonality in automated support requirements.

Our analysis indicates, however, that the great diversity of servicing tasks will often require crew involvement, and teleoperation tends to be more appropriate than full automatic (robotic) operation in many servicing functions, especially when unforeseen conditions are encountered. This contrasts with automation requirements in space-based manufacturing or structural assembly operations where specific functions will be performed over and over, in a systematic, repetitive sequence.

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REPRESENTATIVE AUTOMATION REQUIREMENTS

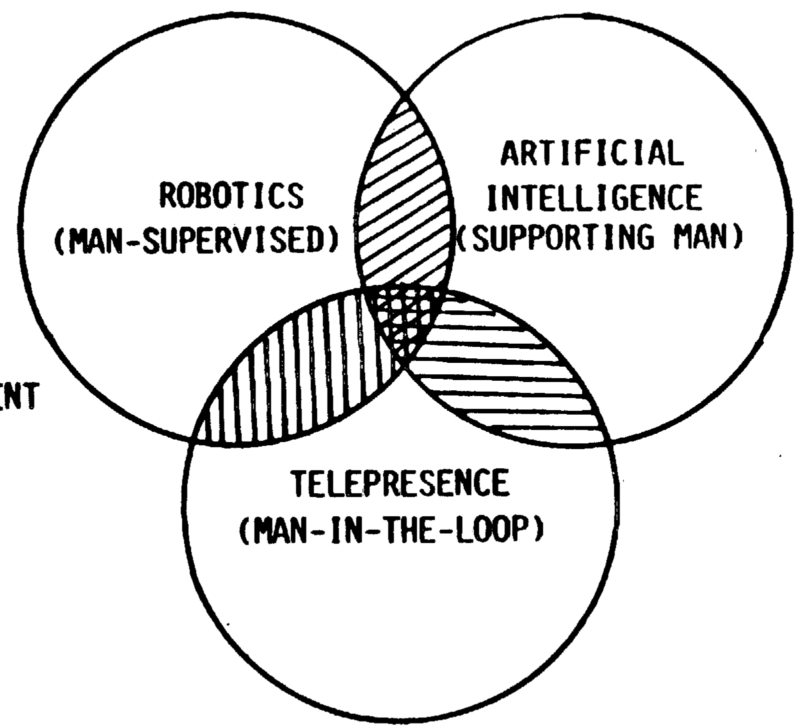
SERVICE FUNCTION \ REQUIRED AUTOMATION SEQUENCE	OPTICAL, MECHANICAL ALIGNMENT, SENSING	ENGAGE/RELEASE MATE/DEMATE	HANDLING, POSITION-ING OBJECT, LOAD	CONTROL TRANSFER DEVICE; PROPULSION	RIGIDIZING	SELECT OBJECT	ACTIVATE, CONTROL SWEEP PATTERN	INSERT, REMOVE SAMPLES, SPECIMENS	CONTROL FLOW TURN ON/TURN OFF	PERFORM TEST SEQUENCE	MONITORING, FAILURE DETECTION	GENERATE WARNING, COMPLETION SIGNAL
LOAD HANDLING, TRANSFER, STOWAGE	✓	✓	✓	✓		✓			✓		✓	✓
ATTACH/DETACH ORU, SATELL. ASSEMBLY	✓	✓	✓	✓	✓	✓				✓		✓
MATING/DEMATING UMBILICAL PLUG	✓	✓	✓		✓						✓	
CONTROL RENDEZVOUS AND DOCKING	✓	✓	✓	✓	✓						✓	✓
FLUID TRANSFER	✓	✓							✓	✓	✓	✓
CLEANING, RESURFACING		✓	✓				✓		✓		✓	✓
CHECKOUT, COUNTDOWN									✓	✓	✓	✓
MATERIALS RESUPPLY, HARVESTING	✓	✓	✓	✓		✓		✓			✓	✓

AUTOMATION DISCIPLINES APPLIED TO SATELLITE SERVICING

The chart shows a logic diagram which defines interrelations between the three principal automation technologies or disciplines used in supporting satellite servicing, and their role in relation to man's functions and tasks. The shaded overlapping areas represent applications that involve joint utilization of more than one of the three technologies, as for example in situations where a remote manipulator is controlled either by teleoperation, with the "man in the loop", or autonomously as robot (usually man-supervised). Teleoperation may be a backup option where robotic use of the manipulator is unable to handle unforeseen aspects of a specific task.

Although not shown in the chart, the Space Station data system plays a major role in providing a critically important link or infrastructure to most or all automated activities

AUTOMATION DISCIPLINES
APPLIED TO SATELLITE SERVICING



- REDUCE CREW INVOLVEMENT
- INCREASES PRECISION IN PARTS HANDLING
- SPEEDS UP SERVICING SCHEDULE

- SUPPORTS TROUBLE SHOOTING
- SAVES CREW TIME
- INCREASES AUTONOMY
- GUIDES SERVICING OPERATIONS

- MINIMIZES EVA TASKS
- ENABLES LOAD HANDLING, TRANSFER
- ENABLES REMOTE (IN-SITU) SERVICING
- ADAPTS TO UNFORESEEN CONDITIONS

TELEOPERATION AND TELEPRESENCE

The chart lists principal areas of teleoperation and telepresence utilization in the performance of satellite servicing tasks and indicates the advantages to be gained by reducing the number of times the crew must perform servicing tasks in the EVA mode. Teleoperation also enables remote (in situ) servicing missions that require direct crew involvement. As an alternative to automated (robotic) servicing operations, teleoperation provides additional flexibility and a potential back-up mode when human intelligence and decision making are needed to handle unforeseen conditions and contingencies.

TELEOPERATION AND TELEPRESENCE*

- TELEOPERATION PERFORMS SERVICING TASKS THAT REQUIRE HUMAN INTELLIGENCE, CONTROL AND DEXTERITY BUT WHICH THE SPACE STATION CREW CANNOT CONDUCT DIRECTLY (E.G., BECAUSE OF EFFORT, SAFETY CONSIDERATIONS, COST).
- IT REDUCES EVA OPERATION REQUIREMENTS.
- IT AVOIDS CREW EXPOSURE TO CONTAMINATION (E.G., IN REFUELING) AND OTHER HAZARDS.
- FREE-FLYING TELEOPERATOR ATTACHED TO AN OMV OR OTV ENABLES REMOTE (IN-SITU) SATELLITE SERVICING UNDER CONTROL OF THE SPACE STATION CREW.
- TELEPRESENCE THUS COMBINES THE ADVANTAGES OF BOTH HUMAN AND MACHINE CAPABILITIES.
- IN REFERENCE SATELLITE SERVICING STUDY BY GRUMMAN AEROSPACE (1981) 60 TO 80 PERCENT OF SERVICING TASKS INVOLVED TELEOPERATION

*CF. "AUTONOMY AND THE HUMAN ELEMENT IN SPACE," STANFORD UNIV., 1 DEC 83

AI (EXPERT SYSTEM) APPLICATIONS TO SPACE-BASED SATELLITE SERVICING

The chart lists principal areas of application of artificial intelligence in support of satellite servicing activities. Expert system support will save crew time and effort, and thus enhance productivity through assistance in planning and execution of complex or intricate task sequences. Of particular concern is the support that expert systems can provide in failure detection, diagnostics, decision making and problem solving, all of which permit greater autonomy of on-orbit servicing from ground support.

AI (EXPERT SYSTEM) APPLICATIONS TO SPACE-BASED SATELLITE SERVICING

- ASSIST CREW IN FAILURE DETECTION, DIAGNOSTICS
- DIRECT TEST PROCEDURES
- CHARTING OF PREFERRED SERVICING MODES, TASK SEQUENCING
- ASSIST CREW IN DECISION MAKING (AUTONOMOUS DECISION MAKING WITHOUT GROUND SUPPORT)
- DIRECT SAFING PROCEDURES PRIOR TO CREW ACCESS
- ASSIST CREW IN RESPONDING TO EMERGENCIES
- PREVENT CREW-OVERLOAD BY MASSIVE DATA FLOW
- LONG-TERM PROSPECT: ENABLE AUTONOMOUS REMOTE SERVICING

ROBOT TECHNOLOGY ADAPTATION TO SPACE STATION USE

The chart addresses the question of which features of currently available terrestrial robotics may be directly applicable to use on the Space Station (left hand column).

The highly developed industrial robot technology provides many features also needed on the Space Station and in satellite servicing such as electro-mechanical design and articulation, computer control, versatility, and programming/teaching principles.

The right hand column lists those issues where major adaptations or modifications are required for robots to work in the new and hostile space environment.

Environmental concerns are primarily those of materials selection, thermal protection, and lubrication techniques. Terrestrial robots generally are designed to work within and exploit the gravity effects that exist on the ground. The design will require modification to operate in zero gravity.

The key issue will be flexibility and adaptability to a great variety of operating conditions and tasks to meet the diversity of satellite servicing functions. Robot applications in space-based manufacturing or structural assembly typically are repetitive in character and therefore would require less flexible designs.

ROBOT TECHNOLOGY ADAPTATION TO SPACE STATION USE

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APPLICABLE KEY FEATURES

- ELECTRO-MECHANICAL DESIGN AND ARTICULATION
- COMPUTER CONTROL CHANNELS
- SENSING TECHNIQUES
- DYNAMIC RESPONSE
- DEXTERITY
- PRECISION
- EXCHANGEABLE END EFFECTORS
- PROGRAMMING/TEACHING ROUTINES

ADAPTATION REQUIREMENTS

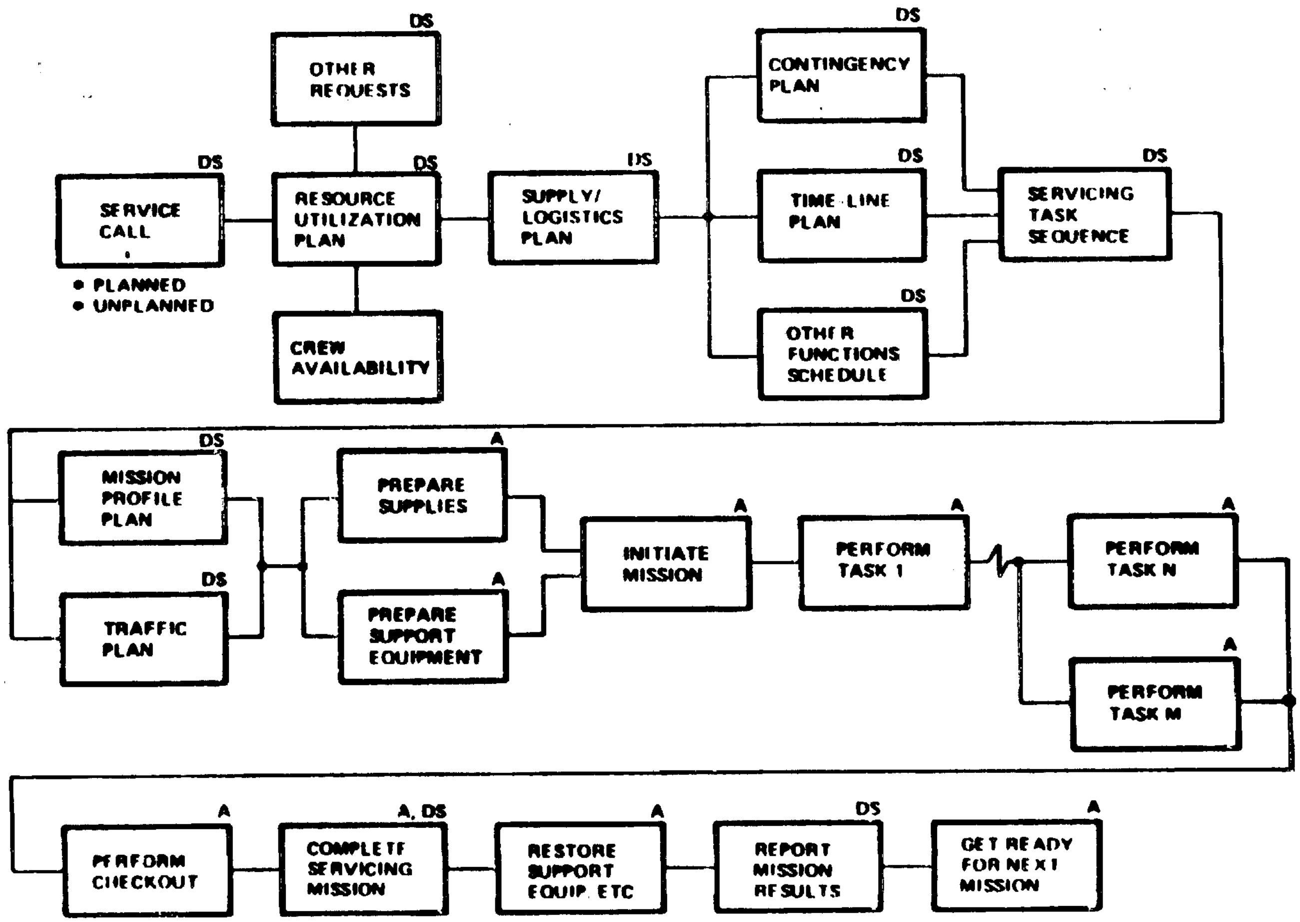
- WEIGHT REDUCTION
- COMPACT LAUNCH CONFIGURATION
- PROTECTION AGAINST SPACE ENVIRONMENT:
  - MATERIALS
  - THERMAL
  - LUBRICATION
- ZERO-g COMPATIBILITY
- ADDED SAFEGUARDS
- OPERATION FLEXIBILITY
- MOBILITY

SERVICING MISSION PLANNING AND EXECUTION

The chart illustrates the important role of Space Station data system support in the planning and execution of a typical servicing mission. The sequence of activities required to perform the mission, starting from the time a call for service is received, is indicated by the flow of major operational steps including resource utilization planning, logistics planning, mission profile planning, preparation of supplies and support equipment through task execution and final checkout.

A large share of these events depends heavily on data system support (indicated by DS). Physical activities involved in carrying out the mission, although not specifically accounted for, are assumed to involve automated equipment support (indicated by A) and often also support by the data system.

Servicing Mission Planning and Execution



A – AUTOMATED SYSTEM SUPPORT DS - DATA SYSTEM SUPPORT

AUTOMATED SERVICING TECHNOLOGY ASSESSMENT

The chart is a first cut at assessing the current state of development of twelve key technologies for the support of satellite servicing. Those required in the earliest servicing missions on the IOC Space Station are expected to be available in the near-term. Many of the technologies for more advanced missions fall in the intermediate category. Longer-term development is needed for items 8, 11, and 12. Knowledge-based systems (or expert systems) will be required to support autonomous, fully robotic servicing functions including automated diagnostics and trouble shooting, and response to contingencies. The reusable orbital transfer vehicle (OTV) will require technology advances to enable in-situ servicing missions to geostationary satellites, not expected to occur before the late 1990s.

The table identifies the listed items as "enabling" or "enhancing" technologies, and ranks priorities on a scale of 1 to 3. Seven of the 12 key technologies have top priority ranking.

AUTOMATED SERVICING TECHNOLOGY ASSESSMENT

KEY TECHNOLOGY	STATE OF TECHNOLOGY: NEAR TERM	INTERMEDIATE	LONGER TERM	ENABLING TECHNOLOGY	ENHANCING TECHNOLOGY	PRIORITY RANKING
1. DEXTEROUS MANIPULATORS, INC. SPECIAL END EFFECTORS	X			X		1
2. SERVICING/AUTOM. SERVICING COMPATIBLE SATELLITES AND PAYLOAD UNITS	X			X		1
3. SPACE-QUALIFIED ROBOTS, ROBOTIC SERVICING		X		X		1
4. DATA SYSTEM SERVICING SUPPORT	X				X	1
5. ADVANCED MAN-MACHINE INTERFACES		X			X	1
6. ADVANCED FLUID TRANSFER SYSTEMS		X		X		1
7. ROBOT-VISION CONTROLLED SERVICING		X		X		1
8. AUTOMATED LOAD HANDLING/TRANSFER			X		X	2
9. AUTOMATED RENDEZVOUS/BERTHING AND PROXIMITY OPERATIONS		X			X	2
10. OMV WITH SMART FRONT END		X		X		2
11. KNOWLEDGE-BASED SYSTEM SUPPORT (TROUBLE SHOOTING, PLANNING, CONTINGENCY RESPONSE			X		X	3
12. REUSABLE OTV			X	X		3

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AUTOMATED SERVICING REFERENCE MISSIONS

As a first step, TRW reviewed the NASA mission model of the 1980s and 1990s to determine likely servicing requirements. However, rather than covering the many projected missions, we concentrated on four representative mission scenarios which encompassed the most relevant aspects of servicing functions to be performed either on board the Space Station itself or remotely at the orbital position of the target satellites. The reference mission scenarios were:

1. Servicing of a low-earth-orbit (LEO) satellite, e.g., the Gamma Ray Observatory (GRO), at the Space Station with orbit transfer by an Orbital Maneuvering Vehicle.

2. Servicing of a free-flying, co-orbiting materials processing facility, in situ, including periodic resupply and harvesting of finished products.

3. Repair/refurbishment or changeout of Space-Station-attached payloads or subsystems.

4. Servicing of a geostationary satellite, in situ, by using a recoverable Orbital Transfer Vehicle to perform the ascent and descent to/from synchronous orbit, carrying supplies, replacement parts, tools and support equipment such as a remote/robotic servicer.

These reference missions are derived from a set of servicing technology development missions (TDMs) previously studied by TRW under NASA/MSFC contract NAS 8-35081 to which this automation study task was subsequently added.

The insert in the center of the chart summarizes principal benefits accruing from automated servicing.

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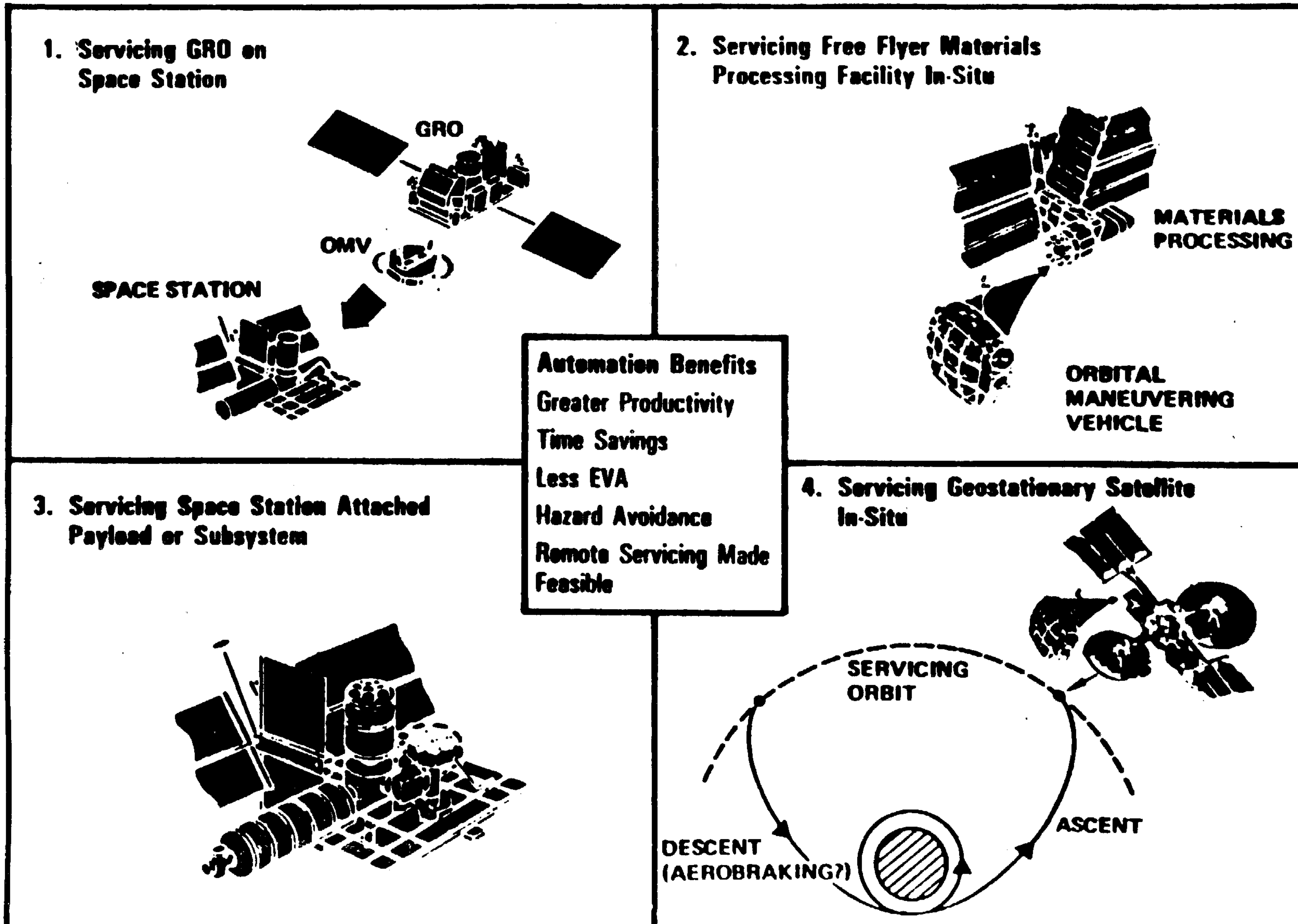
AUTOMATED SERVICING REFERENCE MISSIONS



SPACE STATION DESIGN FEATURES RELATED TO SATELLITE SERVICING
(REFERENCE IOC CONFIGURATION)

We used NASA's current Space Station IOC reference configuration, also known as the "Power Tower," as baseline in selecting a generic satellite servicing concept. The chart shows a simplified drawing of this configuration and indicates areas and features involved with servicing activities. They include satellite storage and service bays, instrument storage, a refueling bay located next to the fuel depot, a bay for accommodating the future OTV and for handling OTV technology development, and storage for the OMV and OMV servicer kits.

A major issue involves servicing on a centralized or decentralized facility. A centralized facility would provide some convenience in terms of close proximity of all servicing related locations and storage areas and would reduce load transfer requirements. A decentralized facility on the other hand reduces local congestion and facilitates growth. It generally emphasizes automated and teleoperated approaches to servicing.

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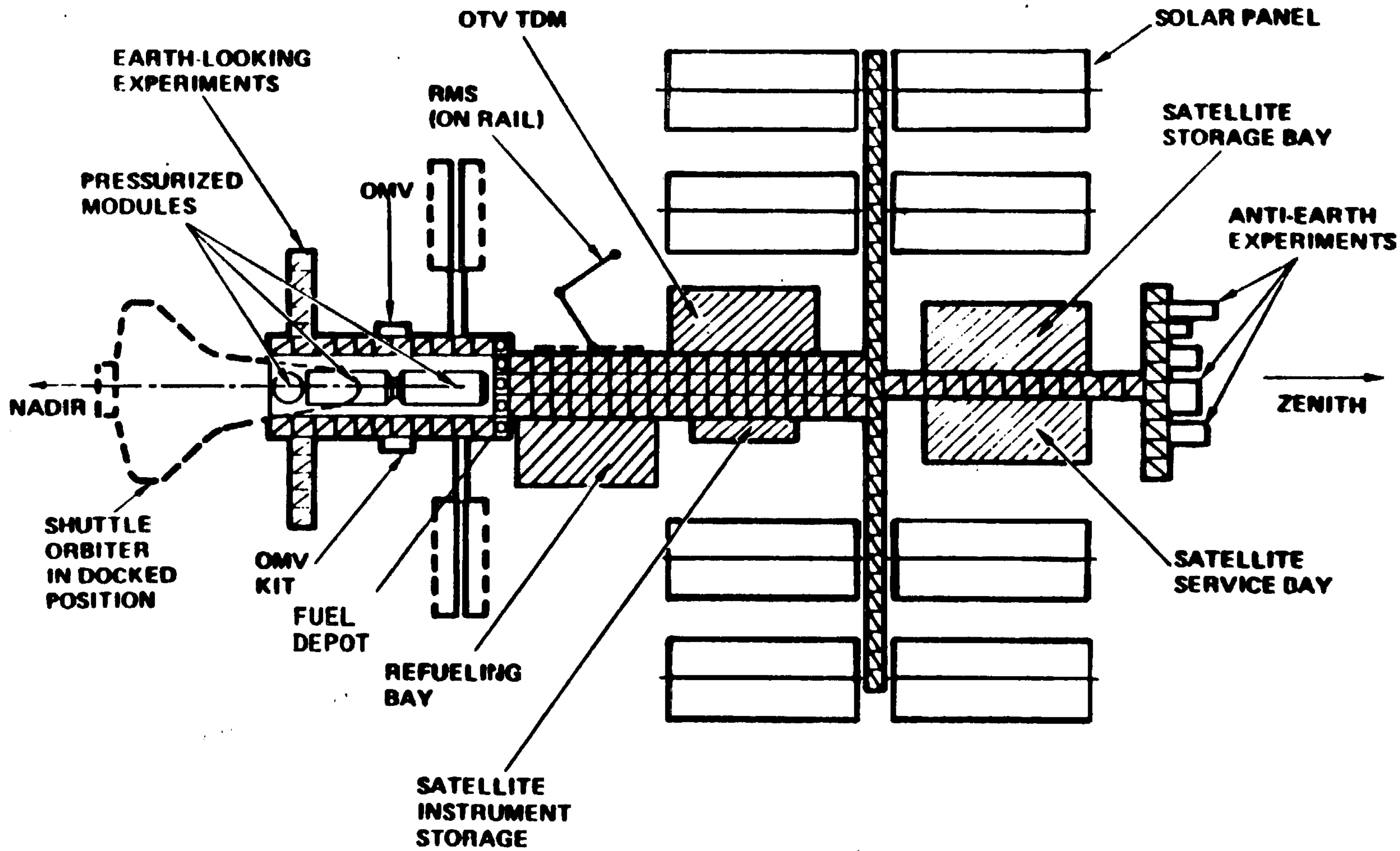
SPACE STATION D. GN FEATURES RELATED TO SATELLITE SERVICING (REFERENCE IOC CONFIGURATION)



TOTAL LENGTH ~400 FT

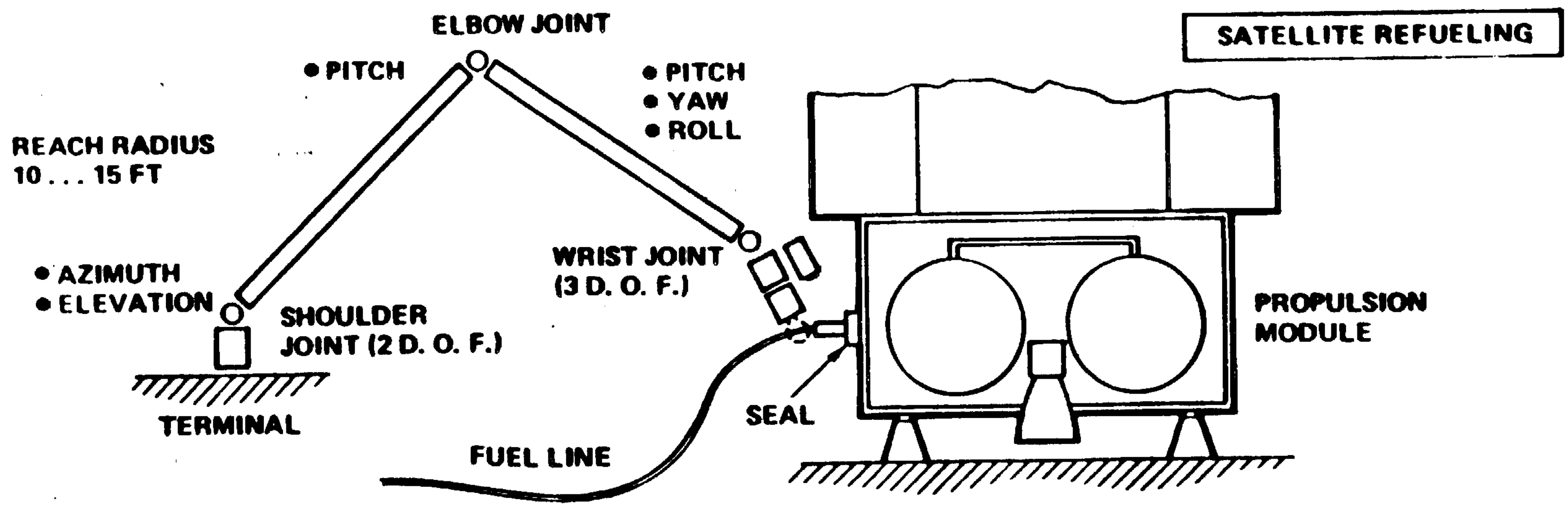
PORTABLE DEXTEROUS MANIPULATOR CONCEPT

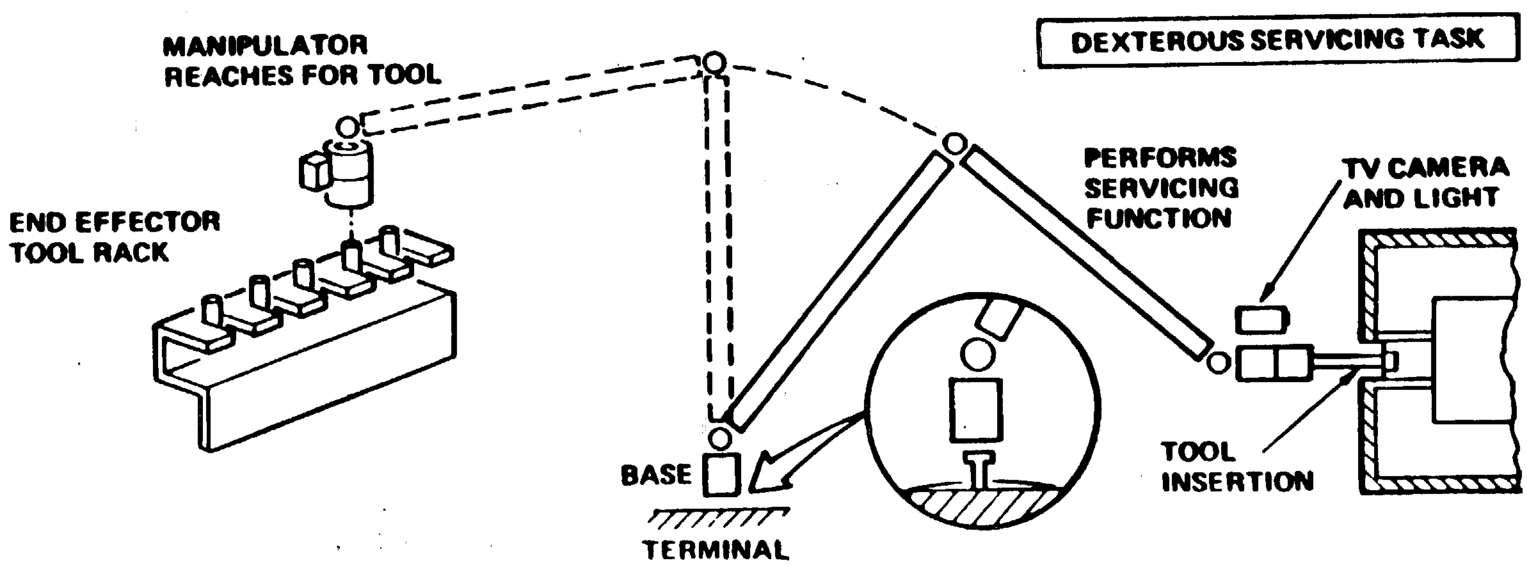
Development of dexterous manipulators is a top priority for most servicinq functions that initially would be performed by hands-on crew operation. The manipulator arm conceivably will have similar articulation as the standard large Shuttle remote manipulator system (RMS), but will only be 1/3 or 1/5 of its size for higher precision, easier control and operation in confined areas. Special end effectors will be the main element in providing greater dexterity. In principle this manipulator will be operational in the man-controlled or robotic mode.

The chart shows two examples of manipulator use, satellite refueling and dexterous tool handling. Automatic changeout of end effectors or tools may be performed comparable to current industrial robot practice.

The manipulator should be designed for portability such that it can be connected to terminals in various locations on the Space Station which provide power and control signals.

PORTABLE DEXTEROUS MANIPULATOR CONCEPT





ROAD MAP FOR SERVICING TECHNOLOGY GROWTH

The chart relates growth in servicing capabilities to the evolution of automation technology. Three major stages of expansion in servicing capability, in the mid '80s, the early '90s and the late '90s, are depicted.

The first stage is limited to Shuttle-based servicing, having been initiated with the repair of the Solar Max Mission spacecraft (SMM) in April 1984 on Shuttle Flight 41C. In addition to actual servicing tasks, the Shuttle also will perform early Technology Development Missions (TDMs).

The second stage starting in 1992 on the early Space Station includes more numerous and more complex servicing missions plus advanced TDMs.

During the third stage, starting in the late 1990s, servicing tasks on or near the Space Station will be performed in a routine manner, repair task complexity will further increase and even geostationary servicing missions may be performed provided the OTV is available with the requisite payload delivery and return capability.

Levels of automation advance from early manual/augmented manual and teleoperation modes through early and advanced robotic modes to near-autonomous modes. The latter incorporate machine intelligence support in diagnostics, troubleshooting, fault isolation and correction, and some levels of decision making.

The earliest milestones in servicing were achieved in three 1984 Shuttle missions, i.e., repair of the SMM spacecraft, fluid transfer demonstration, and retrieval of two communication satellites for repair/refurbishment on the ground. Manual, augmented manual and teleoperation modes were employed with the Shuttle data system providing significant support functions.

As in these pioneering missions any future evolution of servicing technology will require initial phases with men playing a key role in demonstrating and verifying new capabilities.

ROAD MAP FOR SERVICING TECHNOLOGY GROWTH

AUTOMATION LEVEL

1
- MANUAL
- AUGMENTED MANUAL
- TELEOPERATION
- DATA SYSTEM SUPPORT

ABOVE PLUS

2
- ADVANCED TELEOPERATION
- EARLY ROBOTICS

ABOVE PLUS

3
- ADVANCED ROBOTICS
- EARLY EXPERT SYSTEM SUPPORT
- NEAR-AUTONOMOUS OPERATION

SERVICING FUNCTIONS

SHUTTLE-BASED SERVICING (FROM 1984)
- ORU CHANGEOUT
- SIMPLE REPAIRS
- P/L CHANGEOUT
- REFUELING, RESUPPLY
- TECHNOLOGY DEMONSTRATION (TDMs)

EARLY SPACE STATION BASED SERVICING (FROM 1992)
- MORE COMPLEX REPAIRS
- ROBOTS UTILIZED
- REMOTE SERVICING (LEO)
- FREE-FLYING MATERIAL PROC. PLATFORMS SERVICED
- ADVANCED TDMs

GROWTH SPACE STATION SERVICING (LATE 1990s)
- ROUTINE SERVICING ON SS BY ROBOT
- ROUTINE REMOTE SERVICING BY ROBOT
- FIRST GEO SATELLITE SERVICED TELEOPERATION/ROBOT
- COMPLEX REFURBISHMENT AND REPAIR TASKS
- EARLY USE OF MACHINE INTELLIGENCE (DIAGNOSTICS)
- LARGE SERVICING VOLUME

AUTOMATED SERVICING DESIGN REQUIREMENTS

The following two charts summarize principal design characteristics of the Space Station, the Orbital Maneuvering and Transfer Vehicles (OMV, OTV) and satellites which enable or facilitate automated on-orbit servicing.

AUTOMATED SERVICING DESIGN REQUIREMENTS

1. SPACE STATION - PROVIDES:

- INTEGRATED AUTOMATION SUPPORT CAPABILITY BY SPACE STATION DATA SYSTEM WITH DISTRIBUTED ACCESS POINTS FOR
  - COMMANDS
  - DISPLAYS
  - SERVICING TASK SEQUENCING
  - TEST AND CHECKOUT SEQUENCES
- RMS AND TRANSFER SYSTEM TO COVER ALL SS AREAS
- DIRECT LINE-OF-SIGHT COMMUNICATION LINK IN REMOTE SERVICING
- ADVANCED TDRSS DIRECT-LINK SS-TO-SATELLIE COMMUNICATION FOR REMOTE SERVICING
- DEXTEROUS MANIPULATOR, DISTRIBUTED PLUG-IN TERMINALS

2. OMV/OTV - PROVIDES:

- SERVICING KITS FOR TELEOPERATED OR AUTOMATED REMOTE SERVICING
- MULTIPLE TV CAMERAS AND LIGHTING
- CONVENIENT MATING INTERFACES BETWEEN OMV/OTV AND CARGO
- AUTOMATED RENDEZVOUS/DOCKING/BERTHING CAPABILITY

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AUTOMATED SERVICING DESIGN REQUIREMENTS (CONT.)

3. SATELLITES - PROVIDE:

- READY TELEOPERATOR ACCESS TO UNITS EXPECTED TO BE SERVICED
- CONVENIENT REMOVAL/REATTACHMENT OF THERMAL COVERS TO FACILITATE SERVICING ACCESS
- FIXED OR PORTABLE GRAPPLE FIXTURES ON REMOVABLE UNITS
- STANDARDIZED ELECTRICAL AND MECHANICAL INTERFACES ON REPLACEABLE UNITS
- STANDARDIZED FLUID INTERFACES
- REFUELING CAPABILITY
- ASSEMBLY AND DEPLOYMENT CAPABILITY FOR LARGE SATELLITES
- TELEOPERATOR ACCESS FOR DEPLOYMENT/RETRACTION, REPOSITIONING OF APPENDAGES
- EXTERNAL TERMINALS FOR DIAGNOSTICS IN SERVICING AND CHECKOUT.

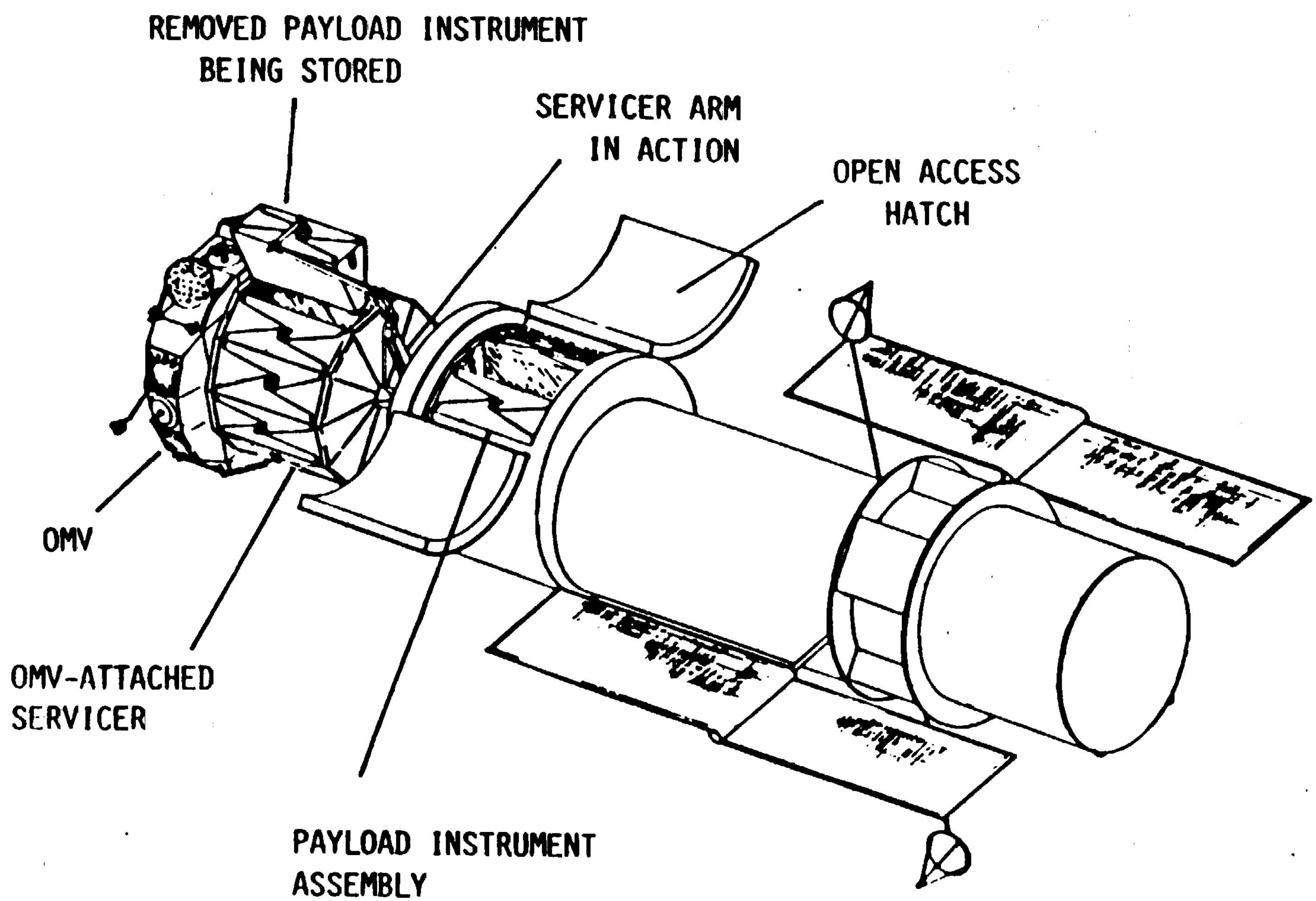
PAYLOAD INSTRUMENT REPLACEMENT ON AXAF

The chart shows an early concept of AXAF being serviced in the free-flying mode by an OMV equipped with a robotic servicer.

The removable payload units are focal plane instruments grouped in a cylindrical arrangement at the aft end of the observatory facility. In the design shown, payload instruments can be removed in radial (lateral) direction. To effect the changeout the servicer, berthed at the aft bulkhead, uses its manipulator arm to reach into the open access hatch, where it pulls out one instrument at a time. The instrument is shown in the process of being stored in an empty compartment of the servicer magazine. The next step for the servicer arm is to take a replacement unit from the magazine and insert it into the AXAF focal plane compartment just vacated.

AXAF servicing is similar to the instrument changeout process on the Space Telescope (HST) however, at this time, neither AXAF nor HST are actually scheduled for in situ servicing, remote from the Space Station.

PAYLOAD INSTRUMENT REPLACEMENT ON AXAF (LATERAL ACCESS)



SOURCE: J. TURNER, "TELEOPERATOR MANEUVERING SYSTEM", SATELLITE SERVICING WORKSHOP, NASA/JSC, JUNE 1982

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CONCLUSIONS

Space Stations are needed to advance on-orbit satellite servicing toward a fuller, more effective and more economical utilization of spaceflight. Starting with a broadened R&D flight program and initial applications in the early 1990s, paced by servicing technology evolution, the objective is to obtain full-scale operation by 2000.

The chart outlines the role of teleoperation, robotics, artificial intelligence and data system support in relation to different mission classes and servicing functions. In-situ servicing in low, and particularly, in geostationary earth orbit becomes a principal driver toward fully automated, robotic manipulation techniques. The Space Station data system will play a key role in providing comprehensive support functions in all phases of satellite servicing.

Twelve automation technologies identified below are key to space servicing.

1. Dexterous manipulators*
2. Servicing-compatible spacecraft*
3. Space-qualified robots, robotic servicing
4. Data system servicing support
5. Advanced man-machine interfaces
6. Advanced fluid transfer systems*
7. Robot vision*
8. Automated load handling/transfer
9. Automated rendezvous/berthing
10. OMV with smart front end*
11. Knowledge-based system support*
12. Reusable OTB.

*Needed for IOC Station

Space-based servicing will draw on current developments in automation technology such as advanced robotics, expert systems, robotic vision, speech recognition, natural language, data processing and display, fault detection/recovery, computing and software. However, practical application of this technology to Space Station automation objectives requires a continuing major development effort. Spin-off benefits to terrestrial applications could be in the area of flexible/adaptable automation, for example in the economical production of small quantities, and in advanced data management and information transfer.

CONCLUSIONS

- MANY SATELLITE SERVICING FUNCTIONS BENEFIT FROM, OR RELY ON, AUTOMATION SUPPORT.
- SATELLITE SERVICING REQUIRES MORE TELEOPERATION AND LESS ROBOTICS THAN OTHER AUTOMATED SPACE STATION ACTIVITIES.
- ROBOTIC SERVICING DEVELOPMENT IS DRIVEN BY IN SITU, PARTICULARLY GEOSTATIONARY, SATELLITE SERVICING OBJECTIVES.
- IN SITU SERVICING BY TELEOPERATION FEASIBLE ONLY IF TRANSMISSION DELAYS ARE REASONABLY SMALL, I.E., 0.25 TO 1.0 SEC.
- MAJOR DATA SYSTEM SUPPORT ESSENTIAL FOR PLANNING, SCHEDULING, EXECUTION, MONITORING AND OTHER SERVICING FUNCTIONS.
- SERVICING SUPPORT BY ARTIFICIAL INTELLIGENCE WILL EXPAND WITH SS EVOLUTION.
- TWELVE KEY AUTOMATION TECHNOLOGIES WERE IDENTIFIED, SOME OF WHICH ARE NEEDED FOR SERVICING ON THE IOC SPACE STATION.
- GROUND-BASED AUTOMATION TECHNOLOGY APPLICABLE TO SATELLITE SERVICING.
- SERVICING AUTOMATION, IN TURN, WILL BENEFIT GROUND APPLICATIONS, I.E., INDUSTRIAL PRODUCTION IN SMALL QUANTITIES, AS A SPACE TECHNOLOGY SPIN-OFF.

RECOMMENDATIONS

In implementing the Space Station Program, NASA intends to advance the state-of-the art in automation and robotics:

(a) for use in Space Station operations, and

(b) to benefit the U.S. economy by exploiting space-based automation progress through technology spin-off to earth-based applications.

In line with these objectives, and based on our study results, we are making five major recommendations with regard to servicing and automation technology as input to the current planning for the Space Station definition phase.

- Crew safety should be the principal concern of defining conventional as well as automated servicing approaches. This requires major attention even in the earliest phases of automated servicing, planning and technology development.

- On-orbit servicing requires that the early and growth Space Stations be designed for rendering effective and economical servicing functions. It also requires that space systems to be serviced incorporate into their configurations, the ability to accept servicing with a minimum of crew effort, support equipment, down time, and cost. This two-way thrust should start as soon as possible under an integrated government (NASA and DoD) policy for designing, planning, and executing of space servicing.

- The IOC Space Station should include automated features such as: load transfer capability, integral verification and test systems, advanced data handling and information processing techniques, a master program for logistics management, appropriate fuel and fluid handling and transfer equipment, and controlled Space Station proximity operations, rendezvous and docking.

- The IOC Space Station must accommodate growth in servicing and automated systems. Provisions for early mods to the IOC Station, through hooks and scars, as well as aggressive planning for expanded resources to support servicing must be reflected in the impending Phase B study efforts and programmatic decisions.

- Key automation technology developments should start as soon as possible. An integrated plan for design, development, test, and evaluation of automation/robotic/AI devices should be formulated and implemented with adequate funding.

RECOMMENDATIONS

- CREW SAFETY IS A PRINCIPAL CRITERION IN DEFINING ALL SERVICING APPROACHES INCLUDING AUTOMATED SERVICING.

- EFFECTIVE SERVICING REQUIRES A TWO-WAY APPROACH:

  (1) SPACE STATION DESIGNED TO PROVIDE SERVICE

  (2) SATELLITES DESIGNED TO FACILITATE BEING SERVICED

  WE NEED TO GET THE CYCLE STARTED.

- THE IOC SPACE STATION SHOULD INCLUDE AUTOMATED FEATURES SUCH AS:
  - LOAD HANDLING, TRANSFER DEVICES AND DEXTEROUS MANIPULATORS
  - BUILT-IN VERIFICATION AND TEST SYSTEMS
  - DATA SYSTEM SERVICING SUPPORT
  - LOGISTICS, SPARES, STORAGE CONTROL
  - FLUID HANDLING AND TRANSFER
  - OTHER

- PROVIDE HOOKS AND SCARS TO THE IOC SPACE STATION FOR EVOLUTIONARY GROWTH ALONG WITH PLANNED RESOURCE CAPACITY FOR EXPANDED SERVICING.

- INITIATE DEVELOPMENT OF KEY AUTOMATION TECHNOLOGIES.

SUMMARY

This chart summarizes highlights, conclusions and recommendations derived from this study. It calls out the major benefits accruing from automated, space-based satellite servicing, indicates the potential technology spin-off to terrestrial applications in the U.S. industry, and emphasizes the need for early funding to initiate the required automation technology R&D effort.

SUMMARY

- AUTOMATION CAN EXPEDITE IOC SPACE STATION SERVICING FUNCTIONS
- TECHNOLOGY EVOLUTION WILL GREATLY EXPAND SERVICING CAPABILITIES
- ORBITAL SERVICING (SATELLITES AND SPACE STATION ITSELF) IS A PRINCIPAL DRIVER OF AUTOMATION TECHNOLOGY DEVELOPMENT
- AUTOMATION ENABLES IN SITU SERVICING MISSIONS
- ASSIGN PRIORITIES TO KEY AUTOMATION TECHNOLOGY DEVELOPMENT AND EARLY SHUTTLE DEMONSTRATIONS
- EARLY FUNDING IS NEEDED TO INITIATE SPACE STATION AUTOMATION R&D
- SOME SPACE SERVICING AUTOMATION TECHNOLOGIES ARE POTENTIALLY TRANSFERABLE TO GROUND "FACTORY OF THE FUTURE" APPLICATIONS.

A JAPANESE EFFORT IN SPACE ROBOTICS

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Abstract

Progress in Space Station Program has been stimulating activities in the field of research and development of space robotics in Japan. This report describes the present status of these activities, such as study on a space manipulator, a teleoperated rendezvous system and R & D of a remote manipulator subsystem for a Japanese experimental module of a space station. And the activity concerning concept study of a free flying teleoperator is also introduced.

1. Introduction

It is reported that so-called industrial robots are widely used in Japanese factories, and the related technology shows rapid progress. These do not imply that Japan is leading the robot technology in general. The Japanese one is specialized in the robots for limited area of mass production. For example, studies on robotics for extremely harsh condition had started recently in Japan. Among them, Japanese technology of space robots is now in the stage of fundamental development. In recent years, the space station project has been activating R & D efforts of space robotics in Japan.

The purpose of this report is to introduce present status of Japanese robotics activities, including research and development, future plans and related technologies.

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2. Study on Space Manipulator at ETL

At Electrotechnical Laboratory(ETL), the study on the space manipulator was started in 1982. Main objective is to establish the basic technology concerning to space manipulators which will be applied to future space stations and space factories.

Following sub themes are set for performing the objective:

i) Space Tribology

The friction and wear properties of the mechanical parts used in the space environment are experimentally studied. The material and lubrication methods of bearings and gears, whose load range are wide for using the manipulator, are investigated.

ii) Actuator

Some types of the actuators employed in space manipulators are developed. The light weight and high reliability are emphasized in design. Brushless DC servo motors with rare earth magnet, a direct drive motor and harmonic drive actuators have been developed.

Fig. 2.1 shows the harmonic drive actuator. This consists of a brushless servo motor, a harmonic gear, an optical incremental encoder, encoder buffer circuits and commutation electronic circuits. Nominal out put of 30 Nm in torque and 0.3 rad/s in rotation rate is obtained, and the weight of 1.2 kg is achieved by light weight design effort. This actuator was operated more than 1000 hours in high vacuum environment of 10^{-7} ~ 10^{-8} Torr. The dry lubricants are coated on bearings and PFPE/PTFE grease is applied to gears.

A high torque actuator of 700 Nm, whose basic design will be succeeded to the Japanese remote manipulator system on the space station, is now under development. Nominal output torque of 700 Nm at the rate of 0.05 rad/s is obtained by the combination of a high torque motor and a novel type of the planetary gear box with the gear ratio of 1/300.

The actuator using the shape memory effect is also

investigated for dexterous operation of an end effector.

iii) Manipulator System

An elbow type manipulator is developed. It can be operated in the space chamber which is evacuated to 10^{-7} Torr or less. The arm has 6 degrees of freedom and a gripper with two parallel fingers. Table 2.1 shows the specification of the manipulator. The arm length is 0.88 m and the tip force is 30 N. The configuration of a developed manipulator is shown in Fig. 2.2. Harmonic drive actuators are used to move the link.

A system block diagram is shown in Fig. 2.3. The manipulator and control/drive unit are set in the vacuum chamber. The control/drive unit includes a microprocessor which generates the trajectory and control the dynamics. The arm has a force/torque sensor at the wrist. The end effector is equipped with optical proximity and interrupt sensors. The grip force is detected by the load cells which are embedded in the finger.

The operator controls the manipulator by the hybrid manner which allows the computer control and manual control. In manual control mode, resolved rate control of translational and rotational motion are conducted by the joy stick in the work space coordinate. Generalized bilateral master-slave control method is studied now.

3. A Simulation Study on Man-in-the-loop Performance of a Teleoperated Rendezvous System at NAL/NASDA

National Aerospace Laboratory(NAL) and National Space Development Agency of Japan(NASDA) have been studying on a teleoperated rendezvous system. A simulation study of the system is accomplished and its summary is presented here[Y. Ohkami, Y. Tadakawa and Y. Yoshimoto].

Objectives:

1) To evaluate performance of a teleoperated rendezvous system

using TV.

2) To define requirements for information display and maneuver system to operator.

3) To define on-board control schemes compatible to this operation.

Assumptions:

1) Human operator on the ground station teleoperates rendezvous vehicle (chaser) based on the TV image transmitted via data relay satellite(TDRSS).

2) Only closing phase is simulated (from approx. 100 m to precontact) between RV and a co-orbiting target vehicle.

3) Target vehicle is co-operative but passive only with a "target cue".

Simulation setup: (Fig. 3.1)

1) Computer Generates Imagery(CGI) system for airplane flight tests at NAL is used for the simulation.

* Image generation - dedicated processor with mini-computer(S/140)
* Dynamics - super-minicomputer(MV 8000)

2) Major information provided for operators.

* TV image with transmission delay(0.1~1.2 sec.) and limited frame rate(20-0.3 frames/sec.).
* Range and range rate with respect to target.
* (optional) Transverse velocities and attitude angles with respect to target frame.

3) Maneuver system: - two sticks.

* Right hand - Attitude rate control
* Left hand - Translation direct control(thruster on-off).

4) Visual Display System Based on CGI Creates a Picture with

* Target vehicle
* Earth
* Background
* Cursors on RV

Major results:

Three airplane pilots conducted the tests and stated -

a. When transmission delay increases from zero to 1.2 sec, RV remains controllable but pilot work load also increases. Each pilot specified his own "optimal" thruster levels.

b. Minimum frame rate was approx. 0.3 frames/sec (one frame every 3 sec.). Sparse frame rate results in increase of flight time and work load since pilot is forced to interpolate the states in-between.

c. Configuration and dimension of the target cue are very important when no other information than TV image is provided to the pilot . Transverse velocities are very useful and in some cases crucial to acceptable performance.

Research Activities -Past and Future-

1982 - Preliminary analysis and software development.

1983 - Simulation tests using miniature spacecraft & TV camera.

1984 - teleoperated system simulation tests using CGI system.

1985 - preliminary analysis of automated/teleoperated system.

1986 - Automated rendezvous system simulation (Phase I).

1987 - Automated rendezvous system simulation (Phase II), and "Rendezvous Flyer" project to be initiated.

4. Research and Development of Remote Manipulator subsystem in Space Station Japanese Experiment Module

Japan is going to join Space Station Program of U.S.. Scientific and technological experiments related to life science, material processing, space astronomy and astrophysics, earth observation, advanced technology and so on will be performed in Japanese Experiment Module (JEM), which is to be attached to the U.S. space station. The JEM consists of a pressurized module, an

experiment logistics module and an exposed work deck. As shown in Fig. 4.1, Remote Manipulator System (RMS) will be utilized on the exposed work deck. The main purpose of the RMS is to support docking of the experiment logistics module with the pressurized module, and to assist experiments on the exposed work deck. NASDA is now playing main role in concept designing of RMS. And a system as shown in Fig. 4.2 is now being studied. The maximum length of the arm is about 10 m which is just enough for manipulation on the work deck of about 4m x 4m, and sufficient for handling the logistic module. The tandem combination of a larger arm and a smaller one is considered now. The larger arm is used for rough positioning and the smaller one is designed to use fine operation. Each of the arms has 6 degree of freedom in motion. The larger one will be driven by resolved rate control mode, and a bilateral master slave control will be adopted for the smaller arm.

R & D of critical components of this manipulator will be done during '85 to '86 fiscal years.

5. Concept Study on Free Flying Teleoperator

Some concept studies on a free flying teleoperator are carried out in Japan. A teleoperator equipped with a laser radar which has a capability to capture and retrieve a satellite with tumbling motion is studied conceptually at Institute of Space and Astronautical Science(ISAS). The study is presented here[I. Nakatani].

The concept is shown in Fig.5.1. The Space Flyer Unit(SFU) is equiped with a laser radar to track corner cube reflectors on the surface of the target satellite. Analyzing the tumbling motion of the target satellite by processing the reflector images of the laser radar using processors boarded on the SFU, the relative position and attitude of the target satellite and SFU are controlled. The SFU and the target satellite are both unmanned and the SFU is controlled semi-automatically and is

operated by a crew on the space station.

The assumed block diagram of the laser radar system is shown in Fig.5.2. Main characteristics are shown in Table 5.1. A conventional GaAs diode with 15 mW power is used as a laser source. A 2 mrad laser beam scans for acquisition of the target satellite. In the small distance phase, a broad laser beam of 30 deg is used and 2-dimensional reflector pattern is detected by a CCD camera and processed. A manipulator with 4 m arm and 5 degrees of freedom is required.

6. Postscript

In Japan, national laboratories and NASDA are now engaged in research and development of space robotics, which are mainly motivated by Space Station Program.

Japanese space projects, depending mainly on unmanned missions, requires development of space robotics as a key technology.

Current Japanese national projects having close relation to space robotics are "Research and Development Programs for Advanced Robot Technology" and "R & D of The 5th Generation Computer". The objective of the former is to develop robot technology applicable for nuclear power plants, undersea operation, rescue operation in disaster area. The results of both projects are expected to be applied to Japanese space robotics around 1995.

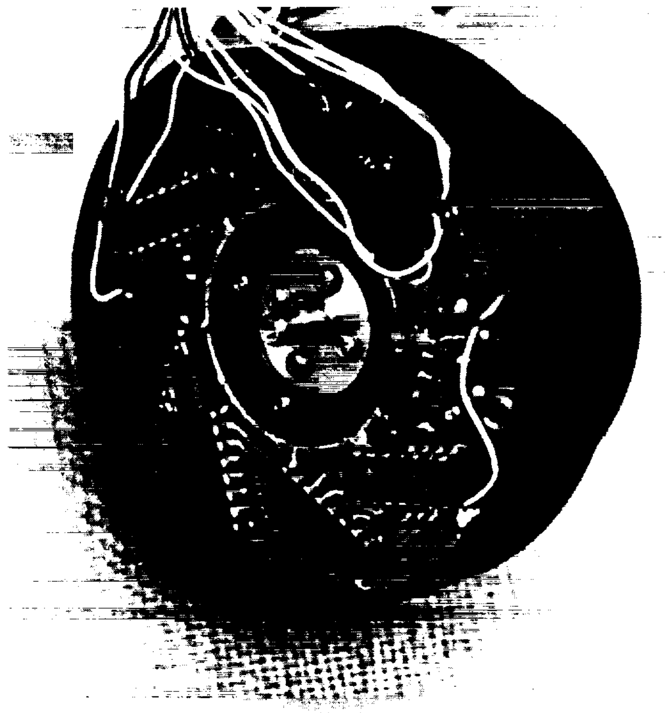

Fig. 2.1 Harmonic drive actuator

Bus voltage	48V
Input power	<140W
Arm degree of freedom	6
Arm length	0.88m
Tip force	30N
Grip force	20N
Control accuracy	
Position (PTP)	±1mm
Rate; Translation	1mm/s
Rotation	1 deg/s
Control mode	
Program; PTP	
Manual; Joy stick resolved rate	
Generalized bilateral	
Arm weight	12kg

Table 2.1 Specification of manipulator system

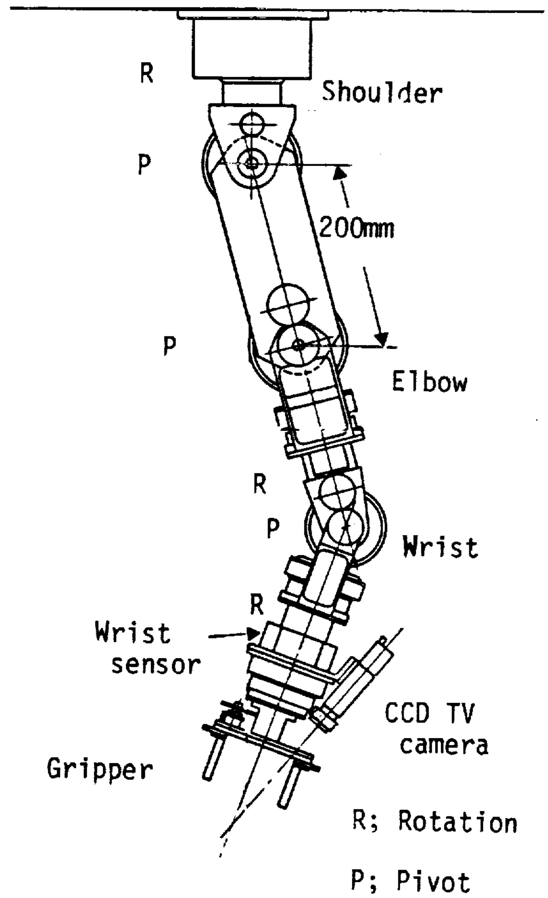

Fig. 2.2 Manipulator arm

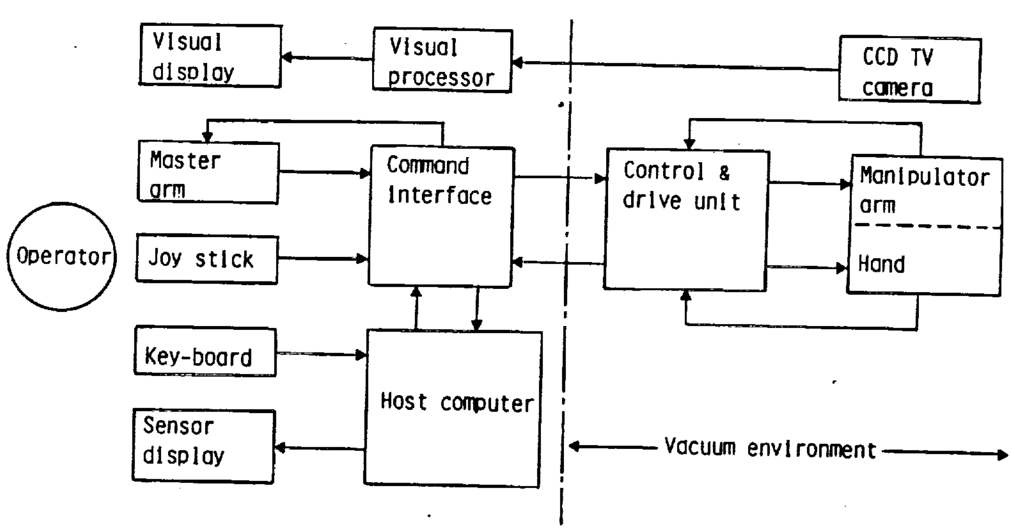

Fig. 2.3 System block diagram of manipulator system

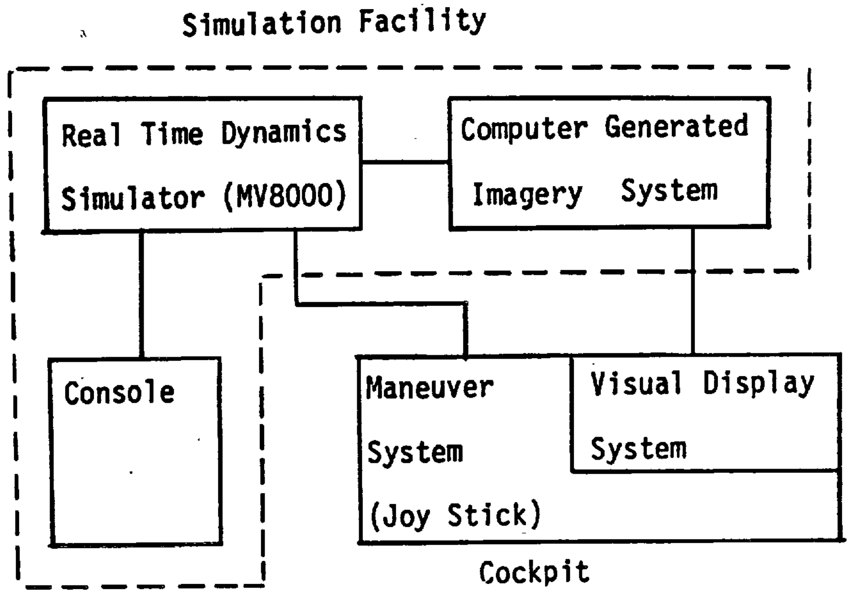

Fig. 3.1 Block diagram of simulation facility

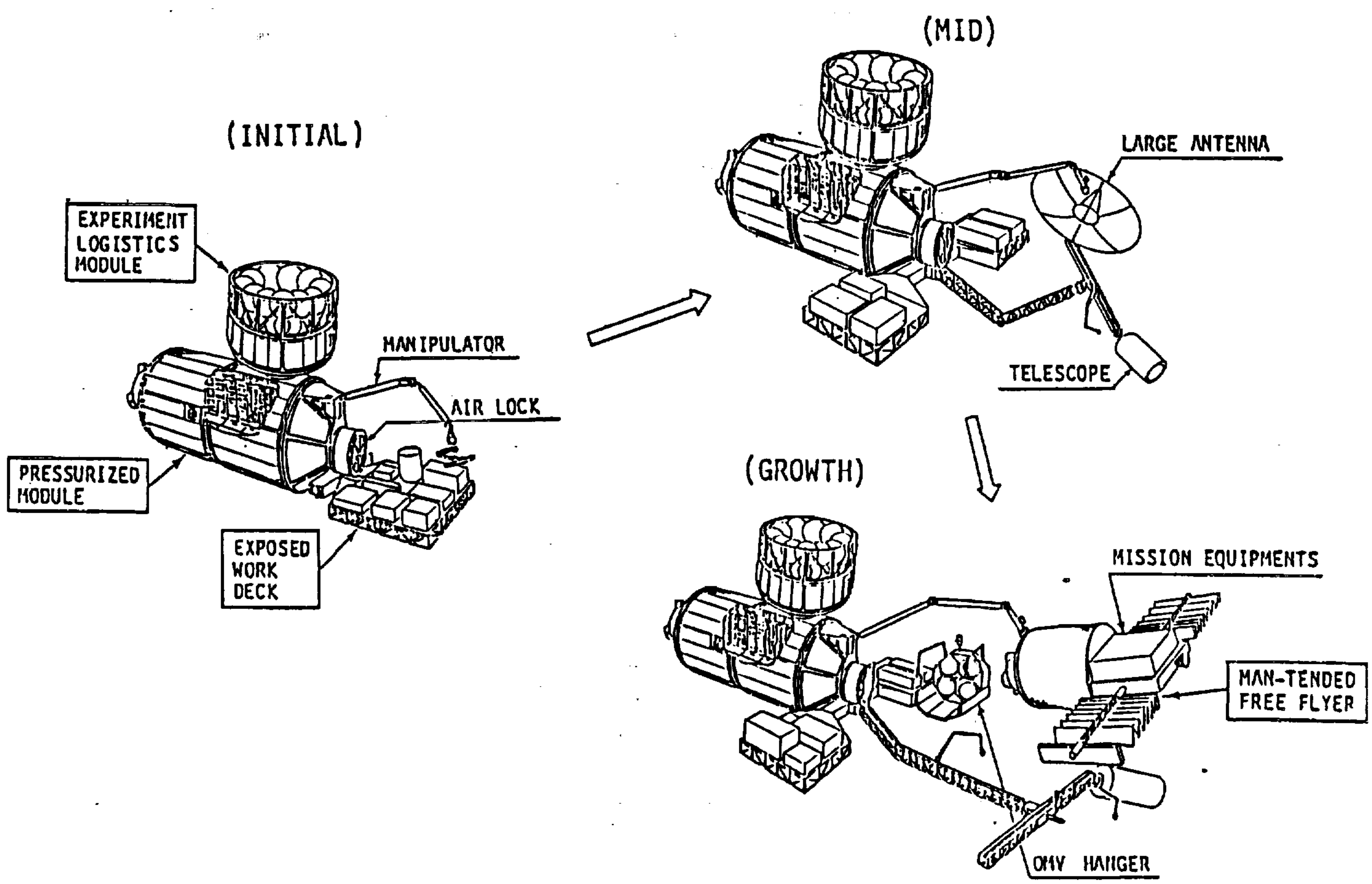

Fig. 4.1 Concept of Japanese experiment module

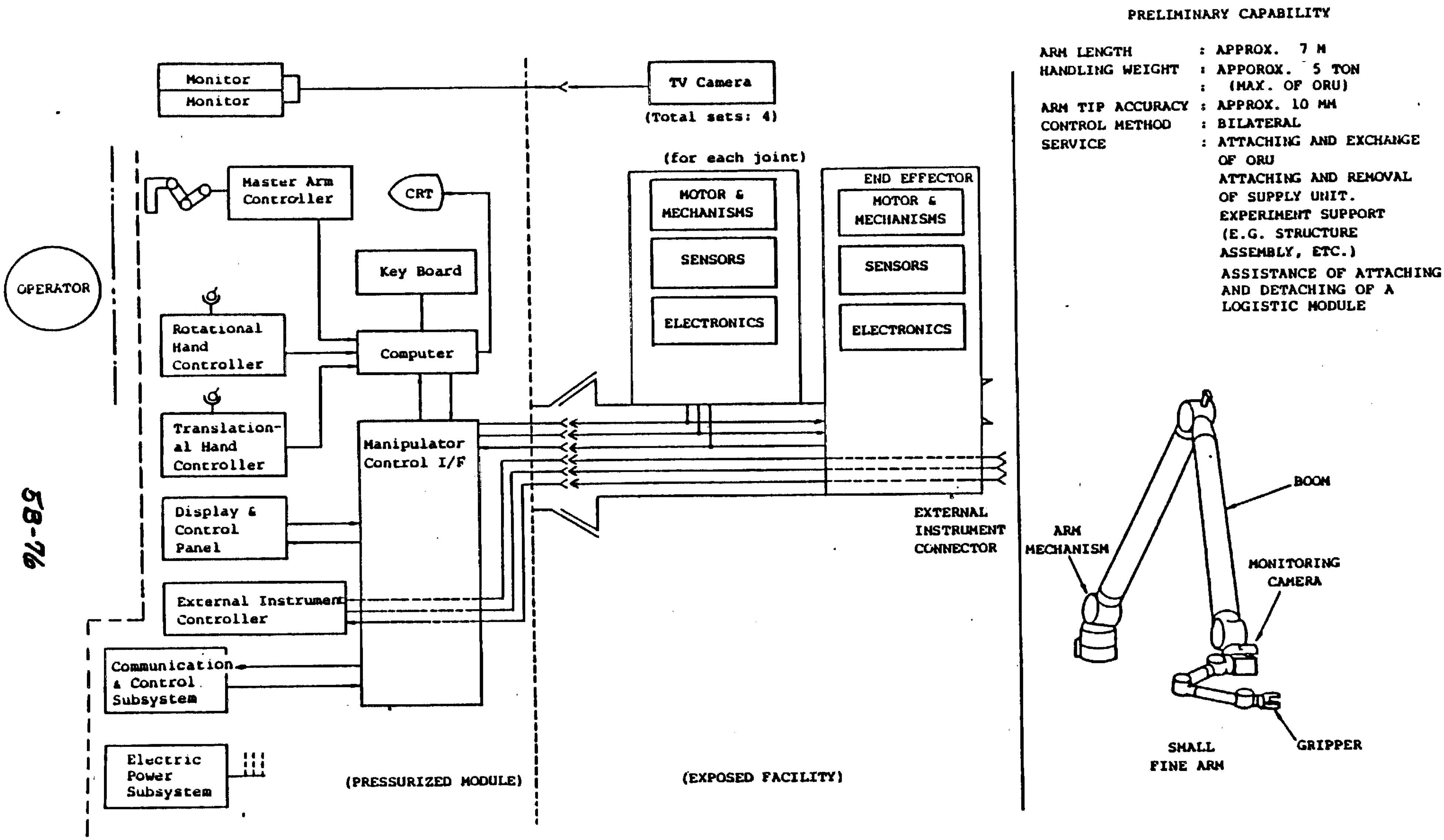

Fig. 4.2 Reference concept of remote manipulator subsystem

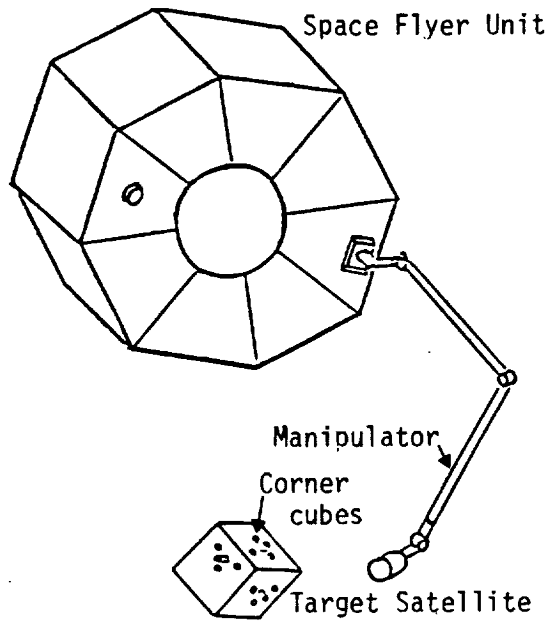

Fig. 5.1 Concept of space flyer unit for satellite retrieval experiment

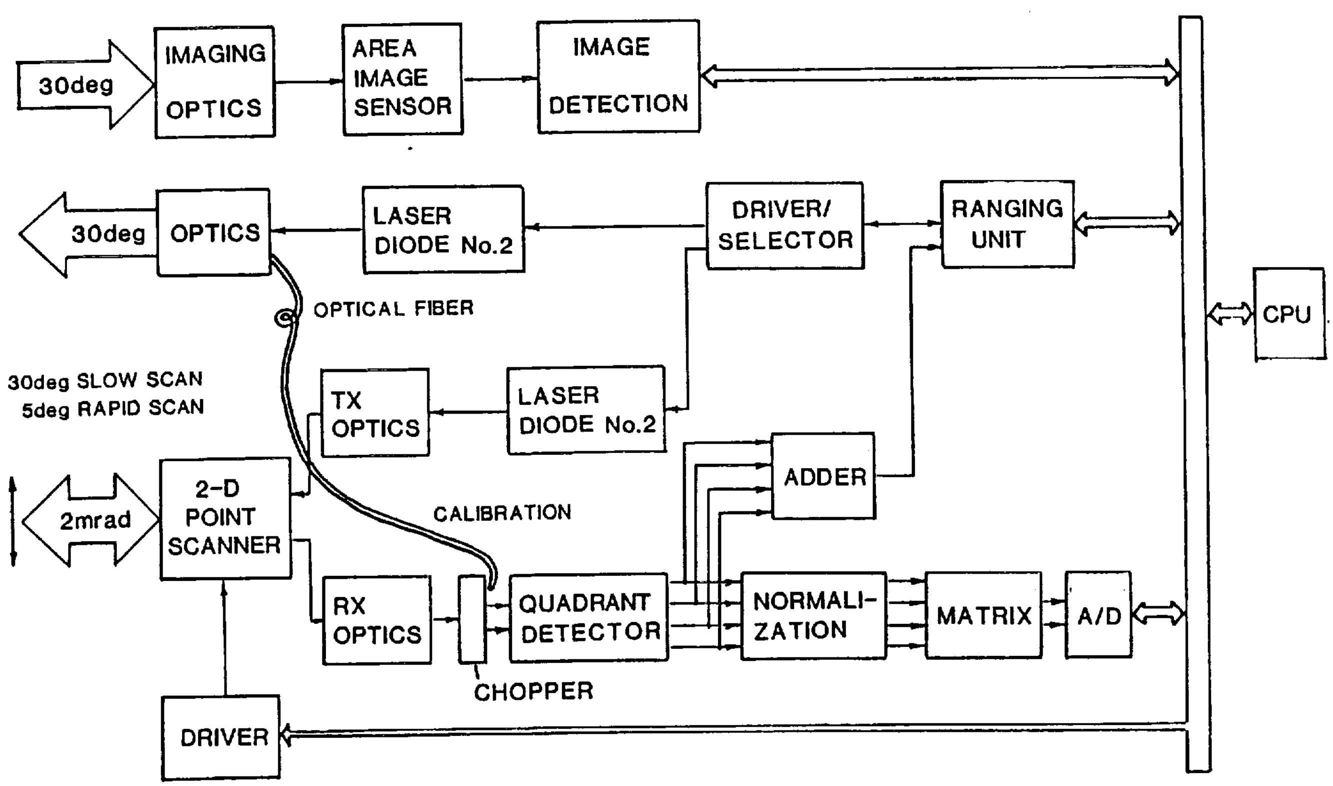

Fig. 5.2 Scanning laser radar system block diagram

	RANGE	MODE	SUB-MODE	LASER BEAM WIDTH		SENSOR BEAM WIDTH		RESOLUTION	SCANNING MODE	SAMPLING PERIOD	TONE FREQUENCY
						CCD	QD				
				0.12°	30°	30°	0.12°				
NEAR FIELD	1m ~200m	TRACKING	TRACKING	—	○	○	—	± 0.08mrad	—	1/30sec	—
			RANGING	○	—	—	○	± 5cm	SLAVE TO CCD	0.1sec	15MHz / 150KHz
FAR FIELD	200m ~ 20km	ACQUISITION		○	—	—	○	2mrad	5°x5° scan	~ 7sec	15MHz / 150KHz / 7.5KHz
									30°x30° slow-scan	~ 250sec	
		TRACKING	TRACKING	○	—	—	○	± 0.4mrad	SLAVE TO QD	0.1sec	
			RANGING	○	—	—	○	± 5cm	〃	0.1sec	

Table 5.1 Scanning laser radar operation mode

TELEPRESENCE SYSTEMS: ANALYSIS AND NEUTRAL BUOYANCY VERIFICATION

David L. Akin

MIT Space Systems Laboratory

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February 19-22, 1985

TELEPRESENCE SYSTEMS:

ANALYSIS AND NEUTRAL BUOYANCY VERIFICATION

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ABSTRACT

Current research conducted by the MIT Space Systems Laboratory (SSL) in space applications of telepresence technology consists of both analytical studies and experimental development efforts. At the conclusion of a two-year effort aimed at identifying space applications of automation, robotics, and machine intelligence systems (ARAMIS), a survey of the current state-of-the-art revealed that all of the component technologies required for an initial operational space teleoperator currently exist, but are scattered in laboratories throughout the U.S. The component requirements of such an initial teleoperator system

are summarized in this paper, and arguments are made for the design goal of the system, which is to be as capable as a current EVA crewman. Along with the requirements of space telepresence systems, capabilities of such systems are determined by detailed examination of candidate satellite servicing tasks, abstracted from those required for current space projects such as Space Telescope and Advanced X-ray Astrophysics Facility.

In a parallel experimental effort, details are presented of the MIT Beam Assembly Teleoperator, used for neutral buoyancy assembly of simulated large space structural elements. This device has been used in extensive simulations, both at MIT and at the Neutral Buoyancy Simulator at NASA Marshall Space Flight Center. Corroborative experiments in manipulator development laboratories at MIT indicate that quality of video feedback is of primary importance in teleoperator productivity, and that force feedback, even in the absense of time delays, is of negligable importance to the particular tasks under study. Free-flight maneuvering of teleoperators is also summarized, through test results of both the BAT and the Multimode Proximity Operations Device, used primarily to examine the role of different sensory modes (kinesthetic, visual, and video cues) used by the human operator.

INTRODUCTION

The term "telepresence", as far as can be determined, was first coined by Marvin Minsky of the MIT Artificial Intelligence Laboratory. The concept is of a teleoperator system with such a high degree of dexterity at the worksite, and with sufficient quantity and quality of feedback to the remote human operator, that the teleoperator system becomes totally "transparent" to the human. Little or no processing power of the human mind has to deal with operating details of the system, and the human can perform the required tasks at the worksite as if he were physically present. Such a system might indeed be said to create in the operator a real sensation of physical presence at the remote worksite.

Clearly, a telepresence system is one ultimate growth goal for any teleoperator system. The other possible growth direction for a teleoperator is toward more autonomy from the operator, through supervisory or fully autonomous systems. It should be strongly emphasized at this point that there is <u>no</u> current meaningful basis for conclusions as to the optimum development path for space teleoperation. Since both telepresence and autonomous systems lie well beyond the current state-of-the-art, there is no way to conclude that

one or the other technology is optimal, beyond the statement of personal prejudices. For this reason, it is strongly suggested that further research into both technologies should progress. In fact, it is entirely possible that the most optimum space worksite in the future might have humans in EVA, telepresence systems, and supervisory control or autonomous robots, all individually working on those task elements which are best suited for each.

TELEPRESENCE ANALYSIS

Between June, 1981, and June, 1983, the MIT Space Systems and Artificial Intelligence Laboratories teamed on a study entitled "Space Applications of Automation, Robotics, and Machine Intelligence Systems (ARAMIS)" for the NASA Marshall Space Flight Center. The first 12-month phase of this study consisted of developing a methodical technique for evaluating potential applications of ARAMIS technology to future space mission.

In order to develop a compendium of tasks to which the ARAMIS technologies could be applied, four space projects were chosen: the Geostationary Platform (GSP), the Advanced X-ray

Astrophysics Facility (AXAF), the Teleoperator Maneuvering System (TMS, now Orbital Maneuvering Vehicle or OMV), and the Science and Applications Space Platform (SASP). Each of these projects was broken down into five successively finer levels of detail. The bottommost level was that of the "functional element", which might consist of a task such as "open access panel" or "position and install new component". The four space projects taken together yielded 327 discrete functional elements. By lumping similar tasks together, this list was simplified to 69 "generic functional elements". For clarity, these 69 tasks were categorized into 9 types:

- Power handling
- Checkout
- Mechanical actuation
- Data handling and communication
- Monitoring and control
- Computation
- Decision and planning
- Fault diagnosis and handling
- Sensing

While all of these tasks could come under some facet of ARAMIS technology, it is obvious that teleoperation and telepresence would be primarily applicable to mechanical handling and sensing.

The next stage of the study involved the definition of potential ARAMIS capabilities. Twenty-eight different fields of ARAMIS were defined, and collected into six areas:

- Mechanisms
- Sensors
- Human-machine interfaces
- Data handling
- Computer intelligence
- Fault detection and handling

At this point, the 69 generic functional elements were cross-referenced to the 28 ARAMIS capabilities. Those matrix points which were not immediately rated "not applicable" were then evaluated on a 1-5 rating scale for each of seven decision criteria:

- Time to complete the task
- Maintenance
- Nonrecurring cost
- Recurring cost
- Failure proneness
- Useful life
- Developmental risk

This cross-reference resulted in a three-dimensional rating matrix with 13,524 elements when fully populated. However,

since the assessment of applicabilities on a 1-5 scale is unavoidably arbitrary, all applicable cross-references of generic functional elements with ARAMIS capabilities were summarized on "capability application forms", which listed both the numerical evaluations with details of background and references for the decisions. There were 465 such forms, which meant that 6-7 ARAMIS capabilities were applicable to each space program task, on the average. While details of this analysis are beyond the scope of this paper, interested readers are referred to NASA CR162079 and CR162083, which together comprise the four-volume final report for this first phase of the study.

Of greater interest in the context of this paper is the second phase of the ARAMIS study, which focused on the applications of telepresence technology to five astronomy satellites:

- Space telescope (ST)
- Advanced X-ray astrophysics facility (AXAF)
- Very large space telescope (VLST)
- Coherent optical system of modular imaging collectors (COSMIC)
- 100m Thinned aperature telescope (TAT)

The advantage of the use of these projects is that they encompassed a list of similar devices, but ranged in time from Space Telescope, which is near completion, to future projects which have yet to be designed. The future projects were conceptual enough to allow the introduction of teleoperation-specific design features, which would compare directly with the EVA-optimized interfaces of the space telescope.

In reviewing the candidate missions, with particular emphasis on Space Telescope, the study identified a list of necessary tasks for an initial satellite servicing system. These tasks consisted of the following:

- Operate mechanical connection
- Operate electrical connection
- Operate latching device
- Grasp object
- Position object
- Operate cutting device
- Operate welding device
- Grapple docking fixture or handhold
- Observe spacecraft/component

The tasks in this list are necessary for NASA to achieve its initial satellite servicing goals, but do not represent a

comprehensive list of the potential capabilities of a telepresence satellite servicer.

Of particular interest is one of the conclusions of the study: the initial goal of an operational space teleoperation system is that the system be capable of the same tasks as current EVA technology. There are several reasons for this conclusion. Current EVA design technology is somewhat conservative in allowing for limited dexterity, mobility, and strength of the EVA subject. These limitations would naturally benefit the design of a telepresence system manipulator and end effector. EVA design methodology is generally well developed, and satellite designers will not be impacted if the presence of a teleoperator system does not require modification of existing design guidelines. Finally, equivalent capabilities of EVA and teleoperation imply that the remote servicer system will have a ready-made market among satellites and vehicles designed for EVA servicing, and that the presence of two complementary servicing techniques will improve the ability for the community to rely on at least one technique being available when necessary.

Based on the development of annotated lists of tasks to be performed in satellite servicing, the study then outlined the

requirements for an initial teleoperation system capable of performing those tasks. The elements of that system include:

- <u>stereo-optic vision system, preferably color, with motion slaved to the operator's head position</u> - this technology currently exists in a few research laboratories. Somewhat limited test results indicate that this is the most natural control mode for a human operator, as well as reducing the dependence on the hands for control actuation.

- <u>Head-mounted visual display system</u> - a natural and neccessary complement to the system above, this would allow the operator to use the video souce, mounted on a pan and tilt unit, in a manner similar to natural head motions for visual surveying of the local area. This technology is becoming more common in research cockpit simulators.

- <u>Two 7 degree-of-freedom manipulator arms with force control</u> - This technology is available in a few production robot manipulators today, as well as in a couple of spaceflight-technology prototypes currently in existence. Such arms, particularly if

anthropomorphic in design, will allow highly natural worksite manipulation in a master-slave mode. Force control allows better master-slave control for working on tasks with constrained motion, or for performing tasks where some effort is necessary for actuation.

- <u>Two grappling arms or one docking device</u> - It is generally a prohibitive use of propellent to react against the attitude-hold mode of the teleoperator mobility base. Instead, it is true of both teleoperators and EVA crew that the optimum work arrangement is where one or more appendages can provide reaction torque against the work site while the primary manipulator(s) perform the necessary tasks. Greater flexibility is gained by having two general-purpose limited-dexterity arms used for grappling available structure in the vicinity, such as EVA hand-rails. Alternatively, attitude torques may be taken out by a specially designed arm with a docking adapter, mating to a specific fixture on the satellite. While this is generally more efficient in terms of mass and torques, it requires a dedicated docking fixture on the spacecraft at each

teleoperator work site, and therefore is not as useful as the general-purpose grappling arm approach.

- Interchangeable or anthropomorphic end effectors - There is a dichotomy of opinions relating to end effector design. It is clear that a variety of actions are required of a telepresence system, similar to those required of an EVA crewperson. One approach is to have interchangeable end effectors, each of which is designed around one of the required tasks. For instance, the end effector could be a powered socket driver for removing bolts, a specialized grabber for removing a module, etc. The contrasting opinion is that, for maximum carry-over of natural control functions, an anthropmorphic dexterous end effector (in short, a robotic hand) would maximize both operator productivity and manipulator dexterity. An operational robotic hand is as yet beyond the state-of-the-art; interchangeable end effector manipulators exist, but have primarily been used in production robotics tasks, and little information exists on human operator adaptation to use of such devices in

master-slave control. More research is clearly indicated, and again the optimal solution may well be a mixture of both end effector stategies.

- <u>Force-indicating hand controllers or exoskeletal master arms for manipulator control</u> - Teleoperated control of manipulator arms has primarily been concentrated in the nuclear industry for handling radioisotopes. Research experience indicates that master-slave manipulation through exoskeletal master arms is a highly productive control mode, although the nuclear industry in the past has used force-reflecting hand controllers almost exclusively. Either technology has been demonstrated workable, although problems yet remain, especially those induced by communications time lag. In fact, some research has indicated that force reflection is contraindicated for significant (greater than .5 second) delays.

The overall conclusion of the study is that an operational space teleoperator, with some elements of telepresence technology, is within the current state-of-the-art as demonstrated by current laboratory hardware. A conceptual

diagram of such a device is shown in Figure 1. With requirements for space-rating and integration (as no laboratory has all the necessary elements for the complete system), such a teleoperator servicer with equivalent capability to current EVA could be available only in the 1992 time frame or later. Those interested in the details of the analysis and the conclusions of the study are referred to the three-volume final report (NASA CR-3734, CR-3735, and CR-3736).

NEUTRAL BUOYANCY VERIFICATION

Beam Assembly Teleoperator

In order to examine the role of teleoperation in space station external operations, the MIT Space Systems Lab has developed a teleoperator system, initially configured for use in structural assembly. This consists of a free-flying neutrally buoyant Beam Assembly Teleoperator (BAT) with two manipulators, and a modular Integrated Control Station (ICS). This system was used in hardware development and check-out tests, concentrating mostly on free-flying operations, grasping, and joint assembly. Future tests will involve complete teleoperated assembly.

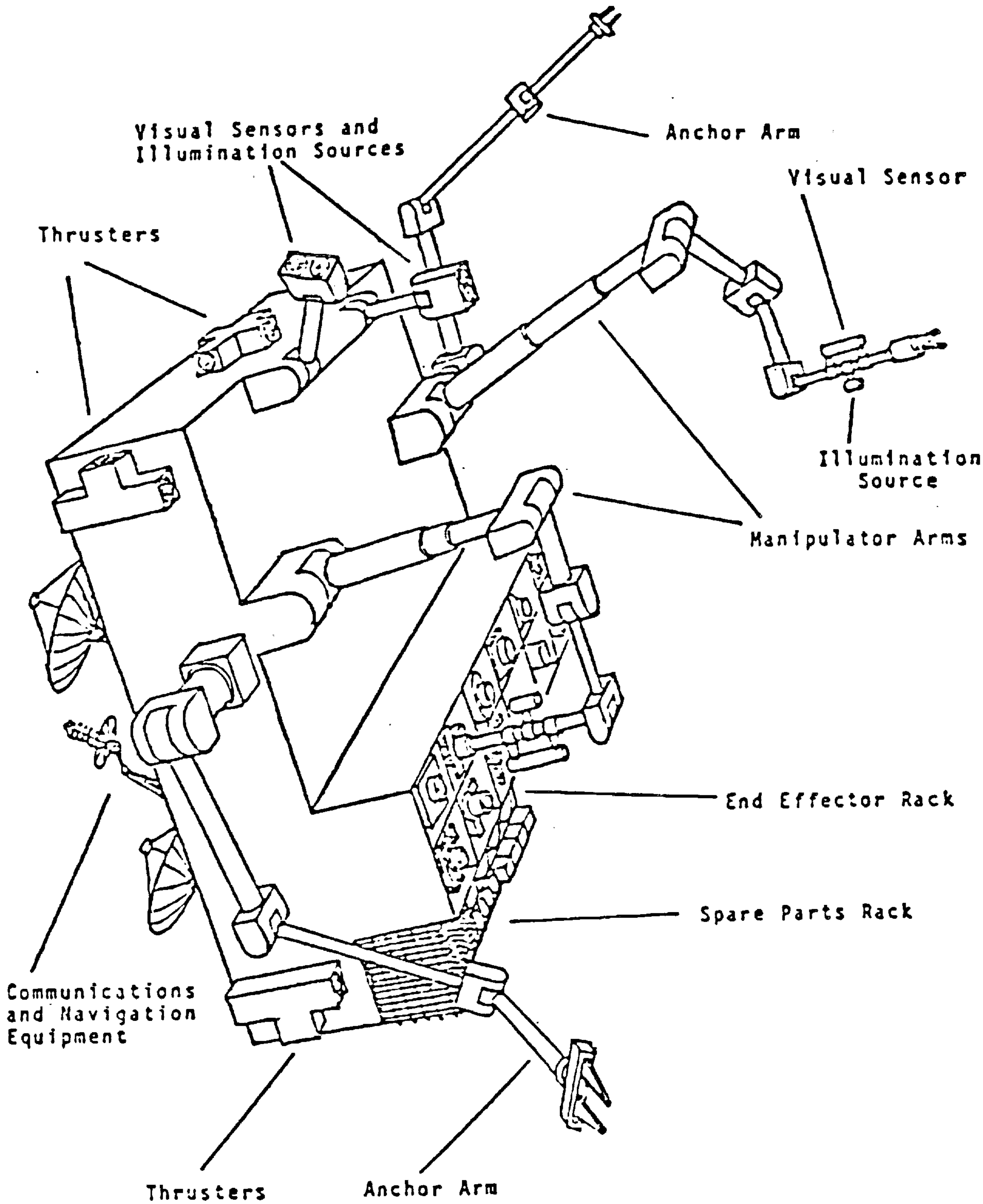

Figure 1: Conceptual Telepresence Servicer Unit

Teleoperation is defined as the use of manipulators and related systems, under the direction of an operator not at the work site, to effect the primary task, such as assembly or satellite servicing. Since a data base of neutral buoyancy simulation of EVA structural assembly had already been compiled by the Space Systems Laboratory, it was decided that the teleoperation system to be developed would be capable of operating in neutral buoyancy, and of assembling the exact same structures used in the EVA tests. Neutral buoyancy allowed the use of unlimited six degrees of freedom, as compared to restricted attitude limits of an air-bearing or motion base simulation. In addition, it allowed the use of arms of sufficient strength and small enough size to perform useful work on reasonable hardware, thus coming closer to the proportions of a realistic space teleoperator.

The basic design of BAT is illustrated in Figure 2. As can be seen, this system is based on a free-flying mobility unit, which uses eight electrical ducted fans to provide all six degrees of freedom underwater. The mobility unit is equipped with one dexterous arm, capable of five degrees of freedom and grasping. This dexterous arm is mounted in the right-arm position, to better accomodate the majority of test subjects. The left arm is a specialized end effector, designed to grasp

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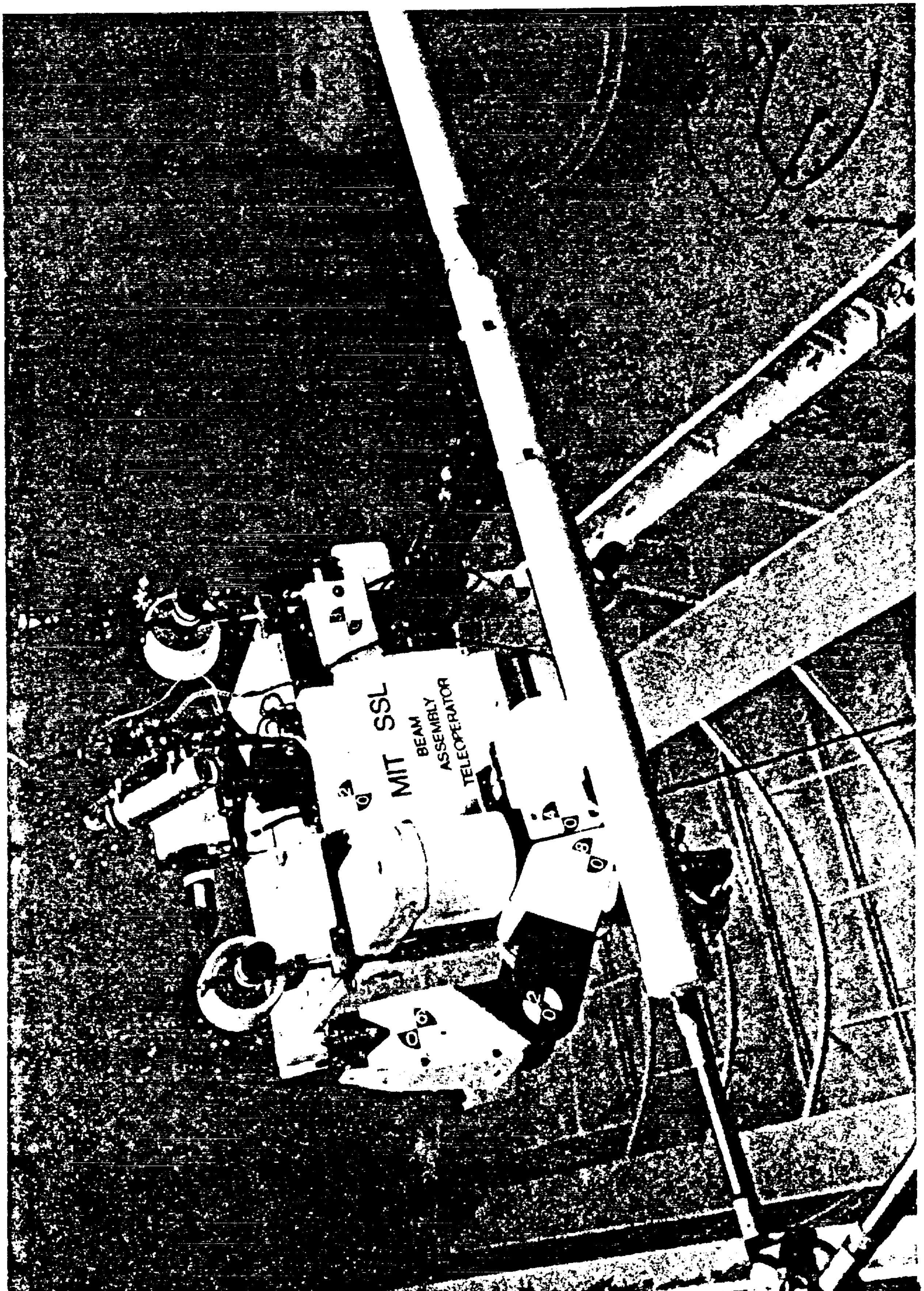

MIT SSL
BEAM
ASSEMBLY
TELEOPERATOR

the 4" diameter cylindrical beam, and, by means of a small drive wheel in the end effector, to move the beam across the body of the BAT along the beam's longitudinal axis. One video camera system is mounted on a pan and tilt unit on the mobility frame, in an anthropocentric head position. A second video camera, later added to the right arm above the shoulder, tracks the end effector. The BAT frame is intended to support some modular interchange of specialized left arms, thus not limiting the teleoperator to the M.I.T. structure only.

The BAT operates from onboard battery power supplies. The main battery box contains the battery power for the arm actuators and the thrusters. The power modules are pulse width modulation amplifiers which drive the manipulator arm's shoulder, elbow, and wrist motors, as well as the camera tilt and pan motors, and the left arm drive.

With the development of the BAT, an equal requirement existed to develop a control station to be used for controlling the BAT in neutral buoyancy activities. To provide mobility, the control station was constructed on top of a 3' by 6' platform truck. Baseline controls for the BAT consist of the following: a master arm on the rear of the control station,

which is kinematically similar to the dexterous arm on the BAT and used to control it in a master-slave arrangement; a helmet with exterior gimbals to monitor head orientation of the operator, and used to slave video camera position to the operator's head position; and two 3 DOF sticks for rate control of the six mobility unit degrees of freedom. Backup switch controls are mounted in the rack panels, as well as six video screens used to monitor the BAT camera outputs, computer graphics displays, and views from the cameras mounted in the shuttle orbiter payload bay mockup at NASA MSFC. This system, shown in Figure 3, is the Integrated Control Station (ICS) as used with BAT.

As an experimental control station, maximum flexibility was desired. Therefore, all controls were interfaced through an IBM Personal Computer, which served as input processor and top-level control system for the BAT. The entire control station is thus software reconfigurable. The subject may elect to use the head controller with either a rack-mounted video monitor, or a 1" color CRT mounted on the helmet to provide monocular feedback.

The BAT control system uses a dynamic model of the teleoperator to compensate for the dynamic coupling of the

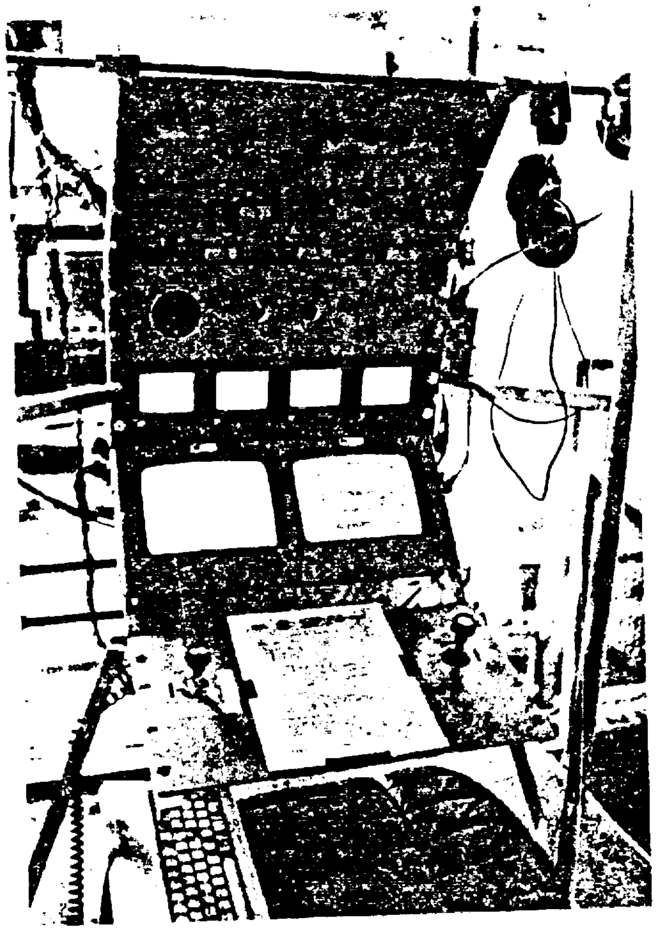


Figure 3 : Integrated Control Station

Telepresence Systems

free-flying frame and the manipulators. Four different control laws were implemented for each actuator and evaluated. The control system is entirely digital and based on three layers of microprocessors in a hierarchical structure. In the lowest layer are joint controllers which maintain closed-loop control of the actuators on the BAT. The middle layer is an intelligent communications buffer. The top layer is a central controller which determines trajectories and transforms between coordinate systems. The power-handling elements of the control system are located on the teleoperator to minimize power transmission losses.

Following systems development testing, the Beam Assembly Teleoperator was tested operationally at the NASA MSFC Neutral Buoyancy Simulator. The views available to the operator in the ICS were the on-board camera of the BAT, a video camera corresponding to the orbital Alpha camera (port side forward bulkhead), and a camera mounted near the surface of the water to starboard of the cargo bay mockup, and looking down on it. These views were continuously shown in the four-inch monitors of the ICS, and could be switched by the operator onto a nine-inch monitor and onto the ICS video-tape system. The video tape thus represents to some

extent the real-time decision of the operator as to the most favorable view for use in operating BAT in the free-flight tests.

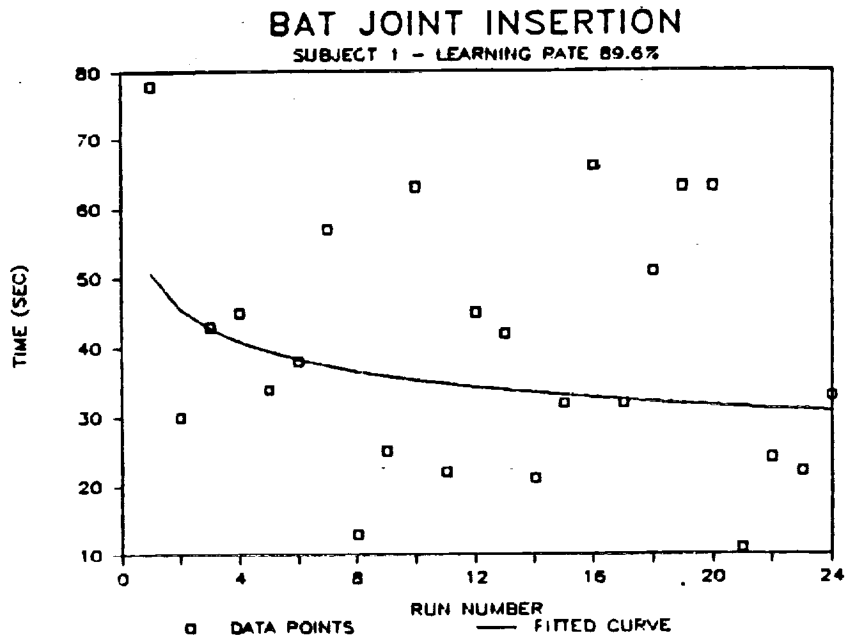
In the MSFC tests, the primary controllers were the two 3 DOF sticks of the ICS. During this test, the sticks could provide input for either the thrusters or the manipulators. The system not currently being driven by the sticks was controlled through single-axis potentiometers. Flight test evaluations showed that the single-axis potentiometers were inadequate for controlling any function, other than simple single-axis maneuvers. The sticks worked very well for input of maneuvering commands, in the traditional form of right hand controlling attitude, and the left hand controlling translation.

In the current configuration, the test subject can control the dexterous right arm through the use of the master arm, which allows natural control of the anthropomorphic dexterous arm of BAT. Further effort is needed to integrate a 6 DOF stick for the control of the motion frame, thus reducing the activities which require the operator to remove their arm from the BAT master arm.

During early free-flight maneuvering tests of the BAT, attitude control authority was found to be brisk, and in fact lead to some pilot-induced oscillations during early learning. This was aggravated by the sensitivity of the mobility frame in all attitude axes, especially roll. In order to reduce this problem and decrease the learning time required for BAT operators, a rate gyro package was procured and will be added to the BAT, in order to provide attitude-hold capability.

Structural joint assembly and additional maneuvering tests have also been performed. Significant operator learning was observed in both maneuvering and manipulation tasks. The presence of two video cameras proved essential to the joint assembly task, as neither camera alone provided sufficient depth cues. Results from the learning studies, where the teleoperator task was the repeated assembly of the MIT locking sleeve structural connector, are shown in Figure 4.

Through the Beam Assembly Teleoperator, the neutral buoyancy environment has been demonstrated as a viable simulation medium for testing concepts of control and dynamics of a space-type teleoperator system. By using a design philosophy which emphasizes modularity and reconfiguration,



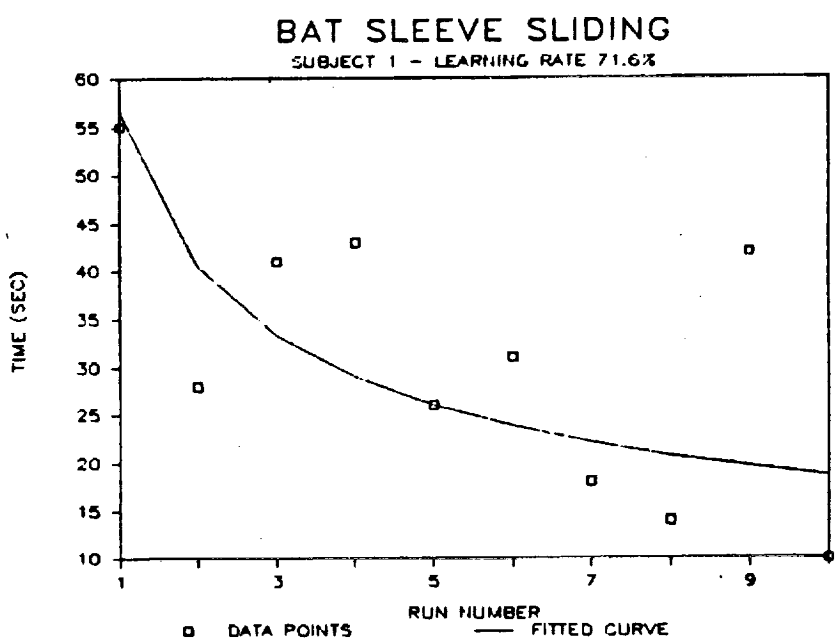


Figure 4 : Learning Curves for BAT

the MIT Space Systems Laboratory has developed a useful test bed for concepts of space teleoperation. The BAT will be tested further performing full-scale structural assembly in neutral buoyancy at the NASA Marshall Space Flight Center during the summer of 1985.

Multimode Proximity Operations Device

While structural assembly is a good test case for many of the manipulative operations around a space station, another critical category is the manipulation of large masses. This will be essential for such operations as payload transfer, assembly of high-energy upper stage vehicles, and indeed for assembly of the station itself. To test these operations, the Space Systems Lab developed a neutral buoyancy version of a generic system that might be used in the vicinity of the space station. The Multimode Proximity Operations Device (MPOD), shown in Figure 5, is designed to perform tasks such as capture and berthing of large masses, and is similar in overall concept to the proposed Orbital Maneuvering Vehicle (OMV), which will perform tasks of this type in close proximity to the space shuttle or space station. The MPOD unit is designed to be used either manned or unmanned: this feature is used to compare manned proximity operations performance (where both visual and kinesthetic cues could be

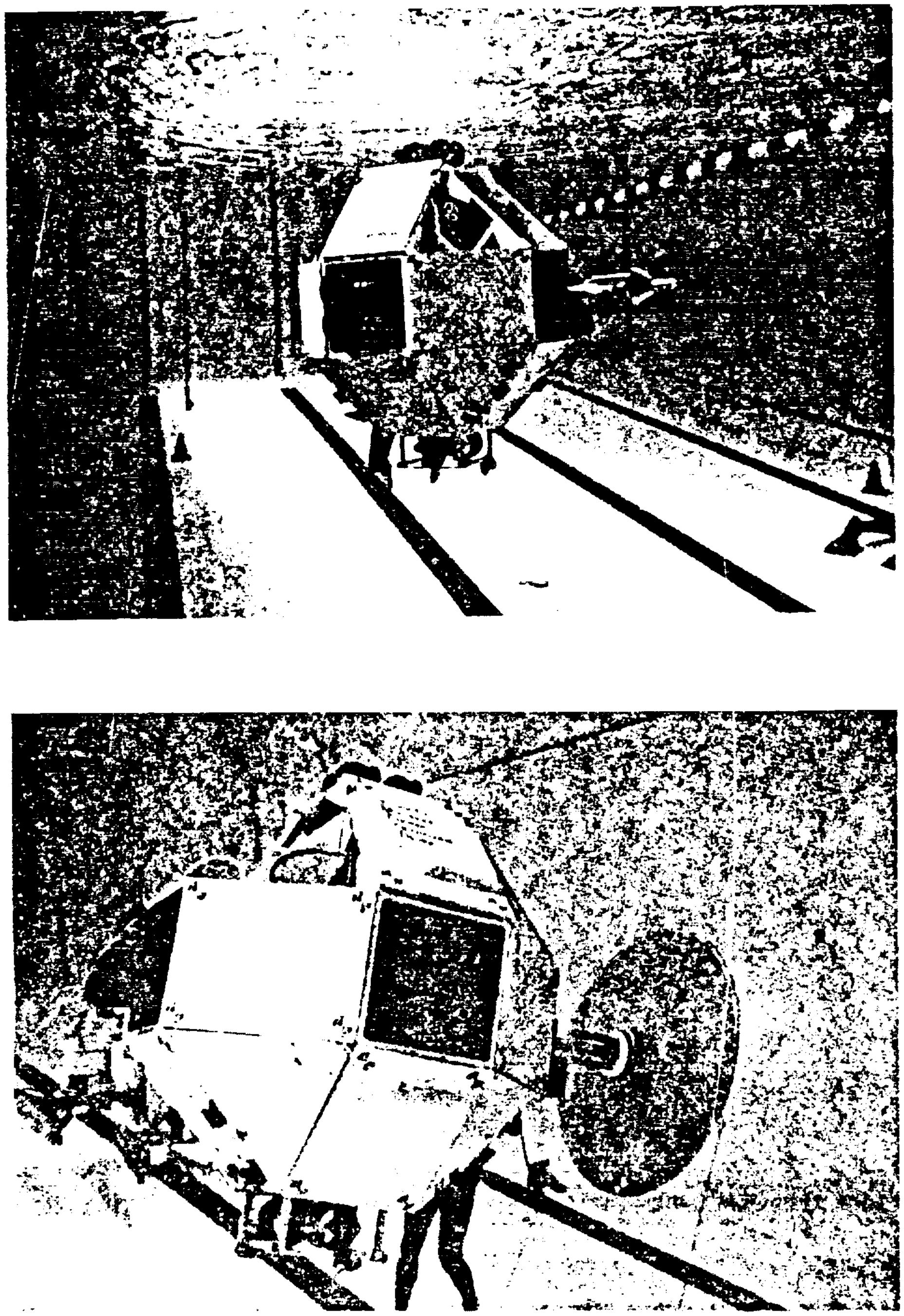


Figure 5: Multimode Proximity Operations Device

Telepresence Systems

used) to teleoperated activities from the space shuttle aft flight deck mockup (with visual cues, but no kinesthetic ones) or from a remote control station (no kinesthetic or direct visual cues). The tasks performed consist of free flight in the close vicinity of the space shuttle payload bay, docking the OMV to a free-flying module (equivalent flight mass of 2100 kg), maneuvering it into position, and berthing it into a restraint fixture in the space shuttle cargo bay mockup.

The MPOD is an enclosed vehicle, with sufficient interior room for a single-person control cockpit. Twelve electric motors, driving ducted and screened propellors, provide six degrees of freedom underwater. The roughly spherical shape produces uniform drag characteristics and simplifies control system development.

Of primary interest in the development of the MPOD was the relative roles of human and machines in space proximity operations. For this reason, MPOD may be controlled either onboard or remotely. The onboard cockpit is of the "wet" variety, with the operator breathing from standard scuba gear while conducting MPOD test operations. For teleoperated activities, a video system can be mounted in the cockpit, and

the vehicle operated remotely with commands transmitted over a serial data link.

MPOD can be fitted with "mission kits" to examine applications to different research areas. In its current configuration, MPOD is fitted with an external docking probe used to grapple and maneuver large masses, and to berth them to pallets on the space station or shuttle orbiter. Another potential research area would be to fit MPOD with a pair of dexterous end effectors, which would also be capable of either direct or teleoperated control. Such a vehicle might be used for operations at an extended distance from a space staion, such as satellite servicing. The potentials illustrate that MPOD is intended to be a general-purpose tool for neutral buoyancy investigations of a wide variety of coming space operational requirements.

SUMMARY

At the end of five years of research, results from the Space Systems Laboratory teleoperator research has shown that teleoperation and telepresence technology will be a viable and desirable operating mode for such tasks as satellite

servicing, structural assembly, and module manipulation. The technology to accomplish this is either already in hand or under research. Of particular note is the list on Page 8 of this report, detailing nine required operations of an initial space teleoperations system. This list was compiled in June of 1983; it is particularly striking to note that by January, 1985, six of those nine required tasks have been repeatedly demonstrated by the Beam Assembly Teleopeator in neutral buoyancy. These technologies will be critical for space station assembly and operations, and further research will allow meaningful allocations of humans and machines for operations in the era of the space station.

ACKNOWLEDGEMENTS

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INCREASED MACHINE INTELLIGENCE AND AUTONOMY

IN SPACE OPERATIONS

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RENDEZVOUS AND PROXIMITY OPERATIONS WORKSHOP

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OVERVIEW

- TECHNICAL BACKGROUND
    - MOTIVATION
    - CONCEPTS AND DEFINITIONS
- MACHINE INTELLIGENCE AND AUTONOMY IN RENDEZVOUS AND PROXIMITY OPERATIONS
    - OPPORTUNITIES
    - TECHNICAL GOALS
- JPL R&D EXAMPLES
    - INTERACTIVE AUTOMATION
    - SENSOR-BASED CONTROL
    - MACHINE VISION
- FUTURE R&D NEEDS AND CHALLENGES

ABSTRACT

The control of rendezvous and docking operations is a complex kinematic and dynamic control task. Development of data driven automation techniques, using machine and task models and on-line sensing will permit the operator intelligent task level supervisory control. The human operator interface of autonomous systems requires hierarchial control system development. The long-term motivation is to make increased utilization of space economically practicable and to reduce human involvement in space operations while increasing systems performance. Broad technical goals are to increase systems intelligence and autonomy through capability to perceive the environment, plan goal-directed behavior, and execute plans while monitoring and verifying machine behavior. Near-term teleoperator and supervisory control technologies which allow man-machine interaction in performing remote tasks with systems having sensor based control and interactive manual-computer control are illustrated. An autonomy thrust at JPL which has used hybrid position/force sensors, computer vision and a robot manipulator to track, grapple, and damp the motion of a free-swinging satellite mock-up is also illustrated.

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TECHNICAL BACKGROUND

-- MOTIVATION --

- TECHNICAL NECESSITY; HANDLING INCREASINGLY COMPLEX SYSTEMS AND TASKS REQUIRES INCREASED AUTOMATION.
- A KEY SPACE STATION DESIGN GUIDELINE EXPLICITLY REQUIRES OPERATIONAL AUTONOMY FOR SPACE STATION FOR REASONS OF
    - EFFICIENCY AND
    - SAFETY
- CONTENT OF U.S. PUBLIC LAW 98-371, JULY 18, 1984, EXPLICITLY REQUIRES IDENTIFICATION OF "SPECIFIC SPACE STATION SYSTEMS WHICH ADVANCE AUTOMATION AND ROBOTIC TECHNOLOGIES, NOT IN USE IN EXISTING SPACECRAFT..." FOR REASONS OF
    - INCREASED SPACE STATION EFFICIENCY AND
    - ENHANCED TECHNICAL/SCIENTIFIC CAPABILITIES LEADING TO MORE PRODUCTIVE INDUSTRIES ON EARTH

MOTIVATION

Activities in space are not possible without automation. As complexity of space systems and tasks increases, so does the need for automation. The state of art in current automation technology needs advancements to enable or to enhance many functions envisioned by future NASA programs. In particular, there is a need to minimize onboard or ground crew involvement in operations which are trivial, mechanizable, or not suited for humans.

Advanced automation will elevate human involvement in space operations to higher functional and operational levels.

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TECHNICAL BACKGROUND

-- CONCEPTS & DEFINITIONS --

- PARADIGM 1: INCREASE AUTONOMY IN SPACE THROUGH ADVANCEMENTS IN AUTOMATION AND ROBOTICS.
- PARADIGM 2: ADVANCES IN AUTOMATION AND ROBOTICS IMPLY USE OF MACHINE INTELLIGENCE CAPABILITIES.
- PARADIGM 3: INCREASED AUTOMATION AND ROBOTICS IN SPACE LEADS TO INCREASED PRODUCTIVITY, RELIABILITY AND SAFETY, LOWER COST, AND ENHANCED FEASIBILITY OR SUITABILITY IN TASK PERFORMANCE.
- PARADIGM 4: GROWTH IN EVOLVING SPACE AUTONOMY CAPABILITIES REQUIRES EVOLVING MAN-MACHINE MIX.

CONCEPTS & DEFINITIONS

Autonomy is an attribute that allows a system/subsystem to sustain goal-directed operation without external intervention for a specified time period within a task domain with given performance requirements.

Automation is the use of machines to control processes under predefined or modeled set of conditions.

Robotics is referred to the study and use of mechanical machines which can move around and/or manipulate objects with some degree of autonomy.

Machine Intelligence is referred to computer mechanization of knowledge and procedures the handling of which normally requires human intelligence. In particular, it is concerned with concepts and methods of symbolic inference by a computer and the symbolic representation of knowledge to be used in making inferences.

Man-Machine Mix allows that control and decision is shared and/or traded between operator and machine (computer). In particular,

> Teleoperation is referred to the study and use of vehicles and/or manipulators which receive instructions from an operator to perform some action at a remote location.
>
> Telepresence signifies a teleoperation situation in which the operator has sufficient cues to "feel present" at the remote location where the machine is working.

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MACHINE INTELLIGENCE & AUTONOMY IN RENDEZVOUS AND PROXIMITY OPERATIONS

-- OPPORTUNITIES --

- NEAR EARTH MISSIONS
  - SPACE STATION & SPACE SHUTTLE
  - OMV, OTV & FREE FLYER
  - SATELLITE SERVICING
  - PATH, TRAFFIC, STATION KEEPING, DOCKING/BERTHING CONTROL
  - VEHICLE OR PLATFORM/STATION EXTREMITIES CONTROL

- DEEP SPACE MISSIONS
  - COMETS AND ASTEROIDS
  - PLANETARY SATELLITES
  - PLATETARY ROVERS
  - EXPLORATION BY REMOTE SENSING
  - EXPLORATION BY SAMPLE GATHERING/ANALYSIS

OPPORTUNITIES

The opportunities in automation technology development for rendezvous and proximity operations for near Earth and deep space missions present both overlaping and differing features. The differing features originate from the different mission goals and from differences in variables of the mission environment. The common characteristics, however, invite a common approach in automation technology development.

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MACHINE INTELLIGENCE & AUTONOMY IN RENDEZVOUS AND PROXIMITY OPERATIONS

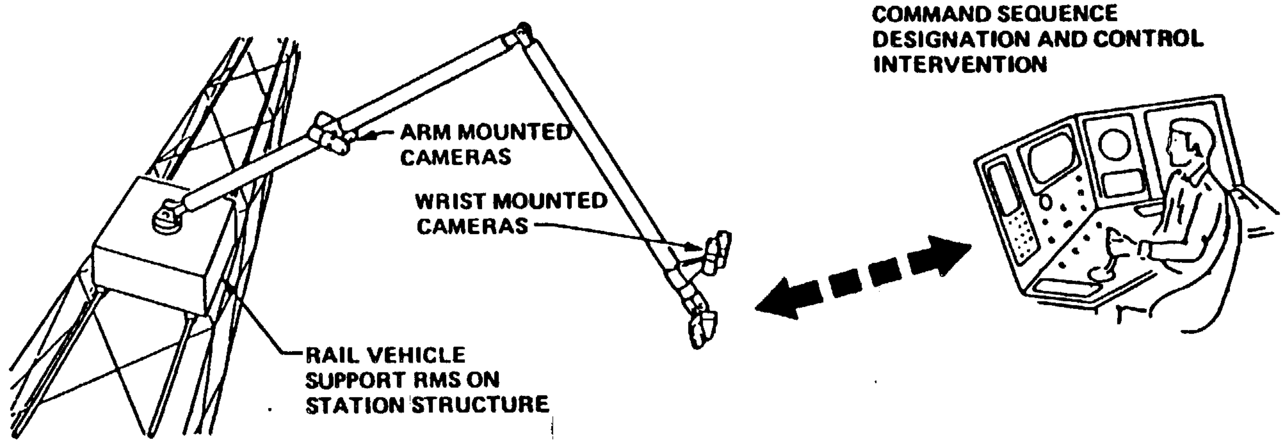
-- NEAR TERM TECHNICAL GOALS --

SENSING AND PERCEPTION

- STEREO CAMERAS
- LABEL/FEATURE-BASED OBJECT RECOGNITION AND TRACKING
- FORCE/TORQUE/POSITION SENSING

SUPERVISED ACTUATION

- HUMAN FACTORS ASSESSMENT OF DISPLAY AND CONTROL INTERFACE REQUIREMENTS
- FUSED SENSOR DISPLAY
- CURSOR INTERACTIVE COMMAND SEQUENCE DESIGNATION AND CONTROL INTERVENTION



INTELLIGENT CONTROL

- PRECOMPILED ROUTINES FOR COMMON SUBTASKS - E.G., THREADING, MATING, FETCHING - STOWING, FASTENING
- ADAPTIVE CONTROL OF LIMBER MANIPULATOR KINEMATICS AND DYNAMICS

TASK PLANNING AND MANAGEMENT

- KNOWLEDGE SYSTEM FOR LOGICAL PLANNING OF SUBTASK EXECUTION SEQUENCES
- AUTOMATED SPATIAL PLANNING OF MANIPULATOR TRAJECTORY IN FAMILIAR ENVIRONMENT
- OPERATOR CUEING AND DIAGNOSTICS FOR ERROR RECOVERY

NEAR TERM TECHNICAL GOALS

First of all, one should take note of the four major thrusts: (1) Sensing and perception; (2) Intelligent control; (3) Task planning and management; (4) Supervised actuation.

The items listed as examples under each thrust area indicate technology goals which seem achievable within three to five years time frame for space applications.

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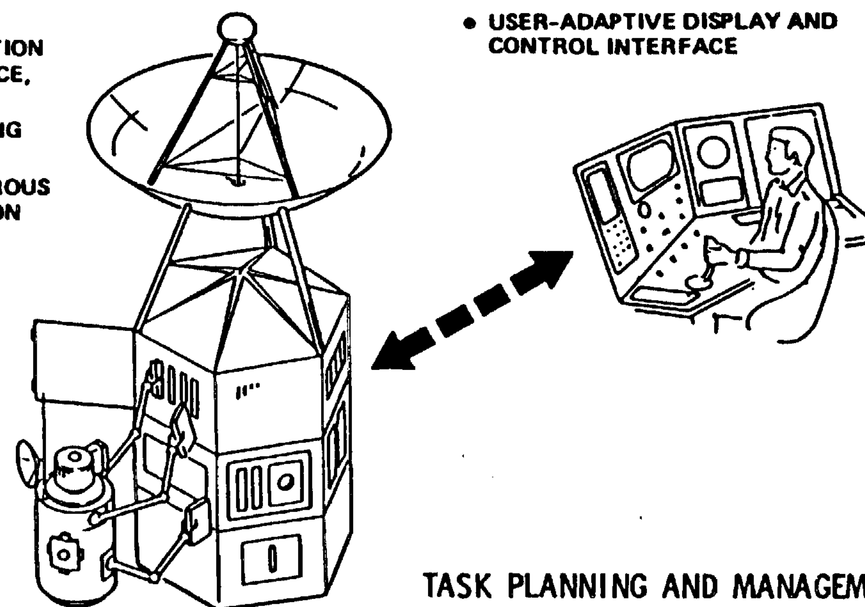
MACHINE INTELLIGENCE & AUTONOMY IN RENDEZVOUS AND PROXIMITY OPERATIONS

-- LONG TERM TECHNICAL GOALS --

SENSING AND PERCEPTION
- COORDINATED MULTISENSING FOR 3-D SCENE PERCEPTION
- VIEW-INDEPENDENT RECOGNITION OF OCCLUDED TOOL, WORKPIECE, AND STRUCTURE
- MODEL-BASED SCENE MATCHING FOR WORKSPACE NAVIGATION
- TACTILE SENSING FOR DEXTEROUS AND COMPLIANT MANIPULATION

SUPERVISED ACTUATION
- VOICE-INTERACTIVE COMMAND
- USER-ADAPTIVE DISPLAY AND CONTROL INTERFACE



INTELLIGENT CONTROL
- SENSOR-BASED SERVO-CONTROL AND VERIFICATION OF TASK EXECUTION STEPS
- CONCURRENT ALGORITHMS AND ARCHITECTURES FOR ON-LINE SYMBOLIC REASONING
- HIERARCHICALLY LAYERED AND MODULAR SUBSYSTEM CONTROLLER FOR FLEXIBLE OPERATOR INTERVENTION

TASK PLANNING AND MANAGEMENT
- EXPERT SYSTEM FOR AUTOMATED TASK PLANNING AND PROGRAMMING
- SPATIAL PLANNER FOR COORDINATED MANEUVER AND MULTI-ARM MANIPULATION IN UNFAMILIAR ENVIRONMENT
- ON-LINE DECISION SUPPORT FOR ERROR RECOGNITION AND FAILURE RECOVERY

LONG TERM TECHNICAL GOALS

The items listed under each technical thrust area indicate (i) increased sensing based intelligence in both information and control and (ii) increased knowledge engineering in task planning and management. A prerequisite for this increase is advancements in computing architectures applicable to space missions.

The items listed under each thrust area indicate technology goals which at some degree seem achievable within five to ten years time frame.

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JPL R&D EXAMPLES IN CONTACT/NEAR CONTACT OPERATIONS

-- INTERACTIVE AUTOMATION --

- **OBJECTIVES**

COMPUTER-BASED → MULTI-DOF MANUAL / SENSOR-GUIDED AUTOMATIC → CONTROLS IN TELEOPERATOR TASK SPACE

- **APPROACH**

DISTRIBUTED MICROCOMPUTER AND MODULAR CONTROL SOFTWARE DEVELOPMENT

- **APPLICATION**

INTEGRATED REAL-TIME CONTROL

MANIPULATORS

END-EFFECTORS

SENSORS

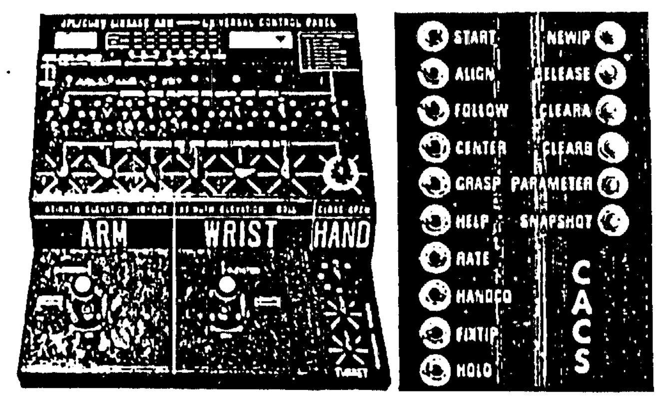
TV SYSTEM

PERIPHERALS

UNDER VARYING TASK CONSTRAINTS INCLUDING TIME DELAYS

COMMAND SOURCES

ANALOG-CONTINUOUS / DIGITAL-DISCRETE



INTERACTIVE AUTOMATION

Interactive automation signifies here a hybrid control capability which permits that some motions of a robot arm in the task space are under manual control while the remaining motions in the same task space are under automatic computer control referenced to information from sensors integrated with the robot hand. The manual control is typically in resolved rate or resolved position mode processed by the computer which also executes the automatic control referenced to sensor data. The automatic control functions are presented to the operator as a task-level control menu. The operator decides on-line when and which automatic control function should be activated or deactivated. The selection of each automatic control function can be accomplished by turning a simple on-off switch addressed directly to the control computer. Some parameters of the automatic control menu can also be changed on-line. In extreme cases all control can be fully manual or fully automatic control.

The test-bed hardware contains a 6 DOF backdrivable hand controller and a PUMA 600 robot arm. The main purpose of this test-bed development is to study, demonstrate and evaluate alternative computing hardware and software architectures required for implementing a hybrid manual-automatic robot arm control system which allows the operator to command and control motions in task-level terms both manually and automatically.

The implementation of this type of control is computation intensive. The development system contains a VAX 730 computer used only for program development. Realtime operation is implemented using a network of dedicated target computers. Programs are developed in C language using UNIX operating system. This permits the development of efficient control programs and the easy transfer of control programs to new systems.

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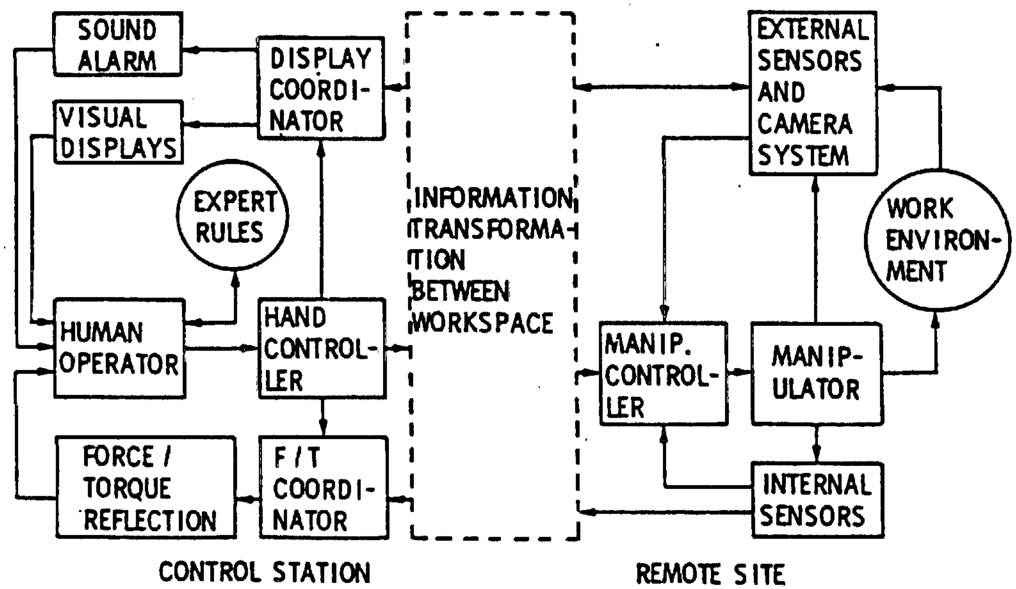
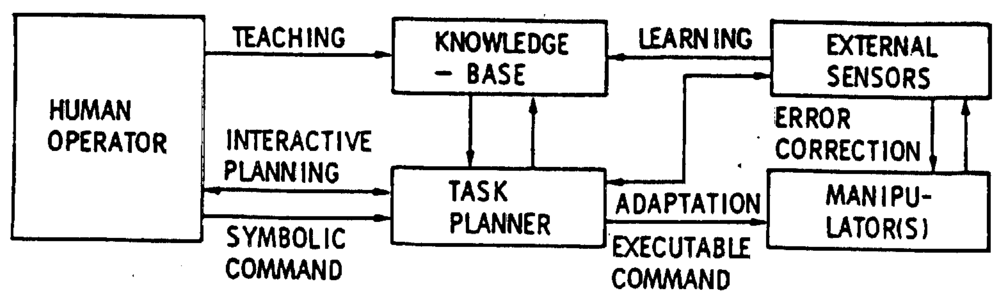
JPL R&D EXAMPLES IN CONTACT/NEAR CONTACT OPERATIONS

--INTERACTIVE AUTOMATION GENERAL IMPLEMENTATION --





INTERACTIVE AUTOMATION GENERAL IMPLEMENTATION

Realization of advanced teleoperators based on generalized bilateral and interactive control requires the solution of a number of theoretical and practical problems, such as

1) establishment of analytic models of generalized bilateral control for the analysis and synthesis of control systems.

2) effective kinematic and dynamic coordination between two workstations in generalized bilateral control.

3) development of sensor-based automatic operations in cooperative/interactive control,

4) cooperative and interactive task planning and

5) effective design of distributed hardware and software systems

In generalized bilateral control, the human operator defines the motion trajectory at a remote site using a hand controller, based on the control situation presented by visual displays, force/torque reflection and alarm signals. The generalized bilateral control emphasizes the design independence between the local and remote workstation, which allows each station to be designed for its own maximum operational effectiveness. The required information transformation and coordination between stations is left to system computers (software linkage); e.g., the kinematics and dynamics of the hand controller may be designed independently of those of the manipulator.

Sensor-based automatic operation carries out the operator's symbolic command (from the function keyboard or voice recognizer) through task planning, error-correction and adaptation, based on system knowledge and sensory information. Using the knowledge-base, the task planner infers an executable sequence of primitive commands in the form of basic manipulator and/or sensor functions. It is noted that task planning may be interrupted by external sensors for adaptation or error-correction, as well as by the operator for interference or assistance.

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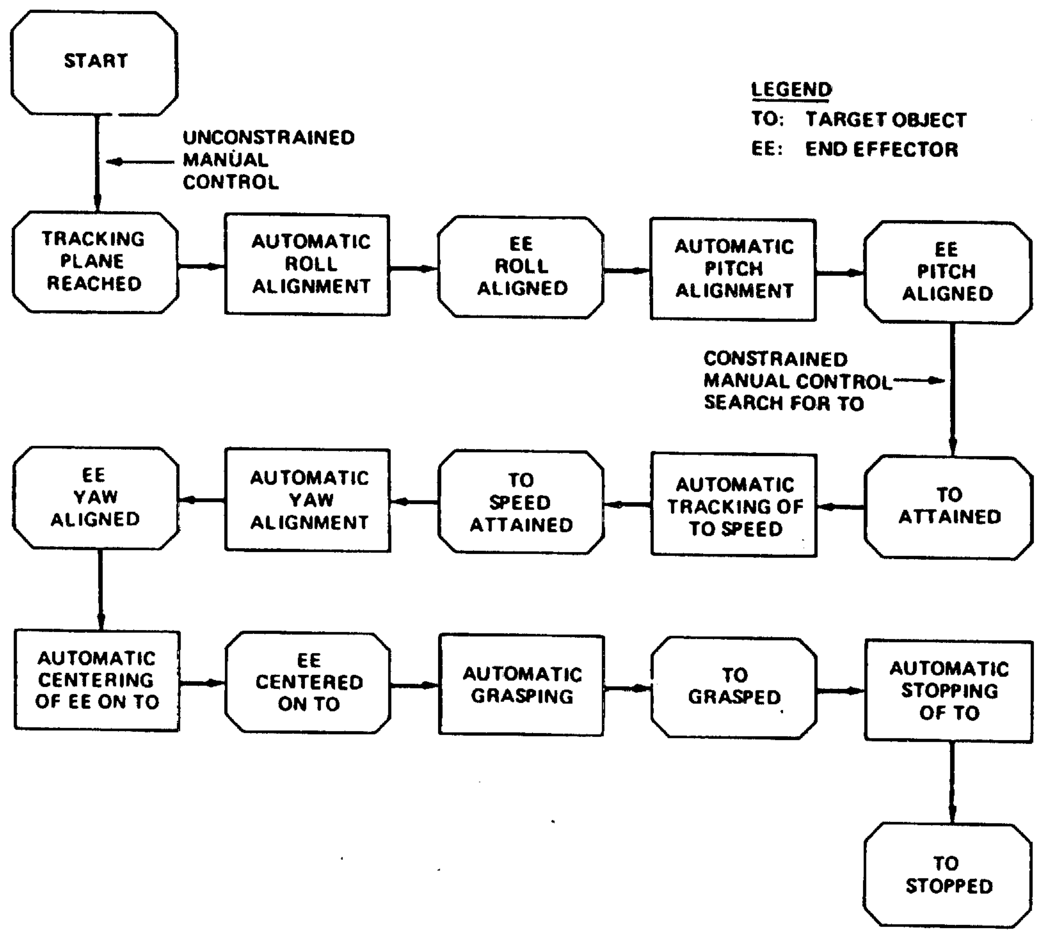
JPL R&D EXAMPLES IN CONTACT/NEAR CONTACT OPERATIONS

-- INTERACTIVE AUTOMATION TASK EXAMPLE --

AUTOMATIC TRACKING/GRASPING CONTROL OPERATIONS SEQUENCE



INTERACTIVE AUTOMATION TASK EXAMPLE

The block diagram shows the interactive manual/automatic operation and system state sequences as they relate to the selected example of tracking and capturing targets moving slowly in a horizontal plane. The operator can select an all-the-way automatic control once the proximity sensors' sensing range has reached the tracking plane under manual control of the manipulator. Or, he can first select any other automatic control action signified by the square boxes in the diagram. After completion of the selected automatic action, the operator can select any other sequentially meaningful automatic operation, or continue the remaining operation manually. In the last case, the system state attained earlier automatically will be maintained automatically during the subsequent manual control for the remaining part of the operation. At any time, the operator can retain full or partial manual control by simple switch turn on/off.

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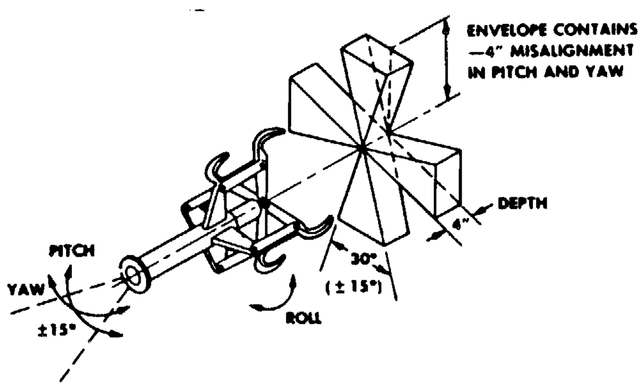
JPL R&D EXAMPLES IN CONTACT/NEAR CONTACT OPERATIONS

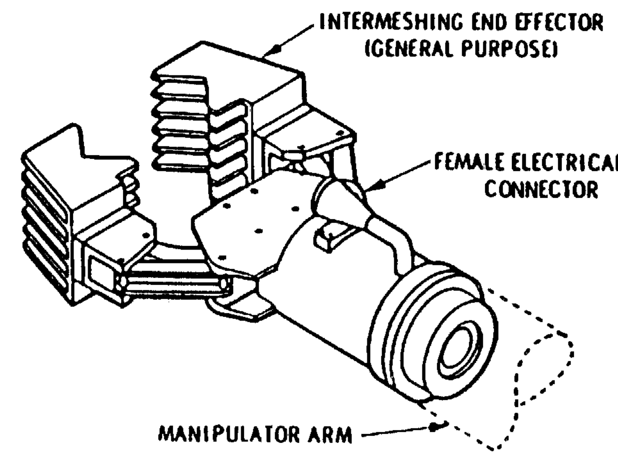
-- SENSOR REFERENCED CONTROL --

SHUTTLE RMS SMART END EFFECTOR



- QUADRANGLE ARRANGEMENT OF FOUR PROXIMITY SENSORS MEASURING THREE STATES IN SIX INCHES RANGE
- FORCE-TORQUE SENSOR AT BASE OF END EFFECTOR MEASURING ALL FORCE AND TORQUE COMPONENTS UP TO 200 LB AND 200 FT-LB.

OMV/TMS SMART END EFFECTOR



- FULL FORCE-TORQUE BALANCE SENSOR AT OUTER END OF WRIST
- GRASP FORCE SENSOR WITH DYNAMIC RANGE FROM 0.25 LB TO 100 LB.
- IN-OUT ARRANGEMENT OF 8 TO 12 PROXIMITY SENSORS MEASURING 4 TO 6 STATES IN THREE INCHES RANGE

SHUTTLE RMS SMART END EFFECTOR

Proximity sensors for Space Shuttle size arm has been developed. This development was aimed to prove concepts and feasibility of sensor-aided control on the scale of a Space Shuttle size manipulator. The sensor system consists of four proximity sensors integrated with a four-claw end effector. The end effector is mounted to a 16-m long manipulator which simulates the function of the Space Shuttle manipulator. The purpose of this sensor system is to aid the operator of the 16-m long manipulator to find the proper depth position and pitch and yaw alignments of the four-claw end effector relative to the grapple fixture of a large payload (e.g., a satellite) before grasping it.

OMV/TMS SMART END EFFECTOR

This end effector integrated with the sensor system will be tested and evaluated on the Protoflight Manipulator Assembly (PFMA) at MSFC in three development phases. The first phase will be completed in CY1985. This end effector has two parallel claws with contoured inner surface.

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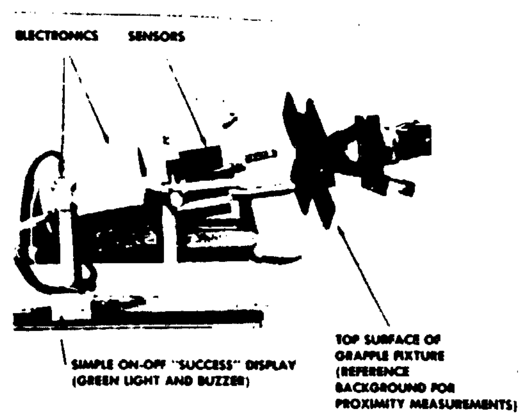
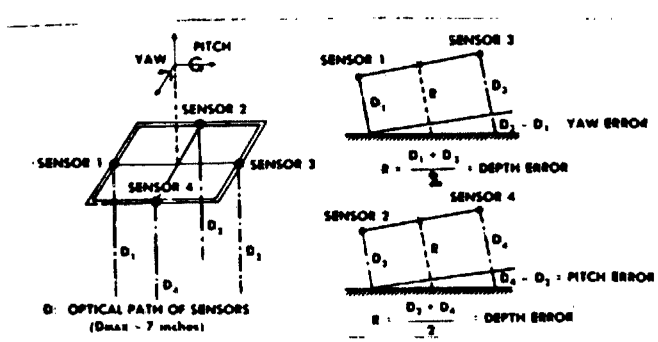
JPL R&D EXAMPLES IN CONTACT/NEAR CONTACT OPERATIONS

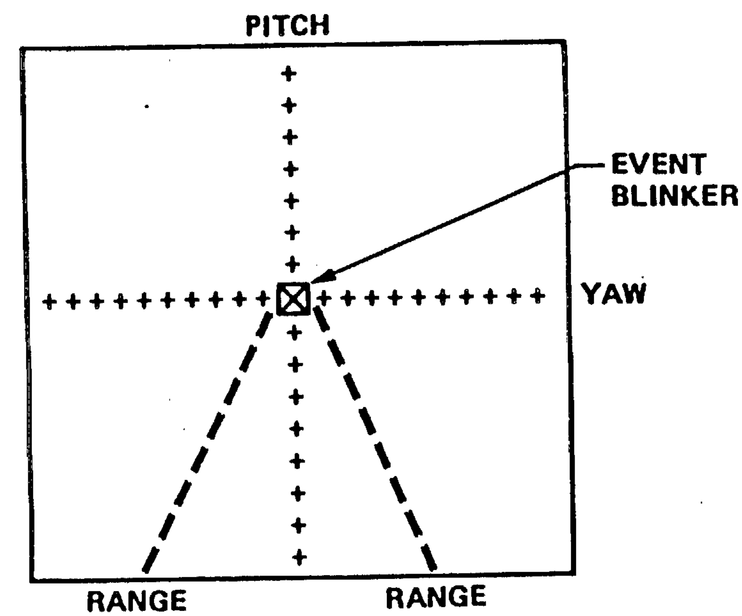
-- SENSOR REFERENCED CONTROL --

SENSING/MEASUREMENT SYSTEM





COMPUTER-DRIVEN INTELLIGENT DISPLAY



PROXIMITY SENSORS ON FOUR-CLAW END EFFECTOR

The sensor system and measurement definitions together with the corresponding information display are shown here. The display shows the operator the values of depth, pitch and yaw errors referenced to end effector axes, in addition to indicating whether the combination of these three errors will assure a successful grasp of the grapple fixture. Showing the values of depth, pitch and yaw errors will aid the operator to fine-control the grasp with respect to these three errors within the grasp envelope.

The graphic display also contains a tone generator. It provides a "success tone" when the center green lamp is on. It also provides a "warning tone" (a short beep tone, distinguished in frequency from the "success tone") when the target reaches the sensing range or leaves the sensing range.

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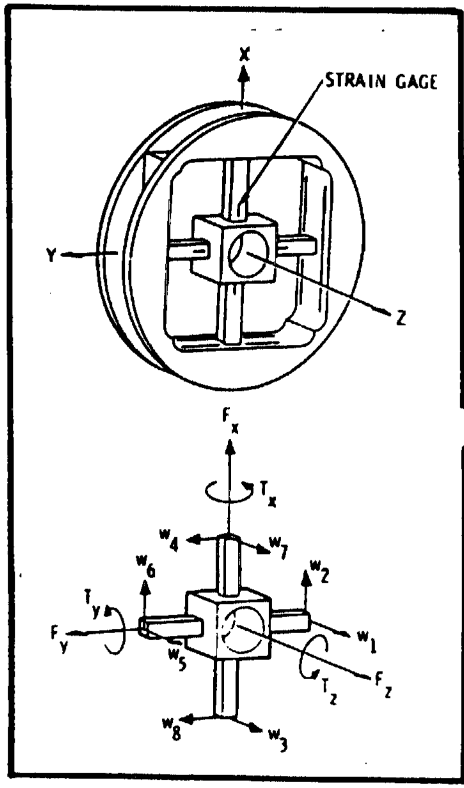
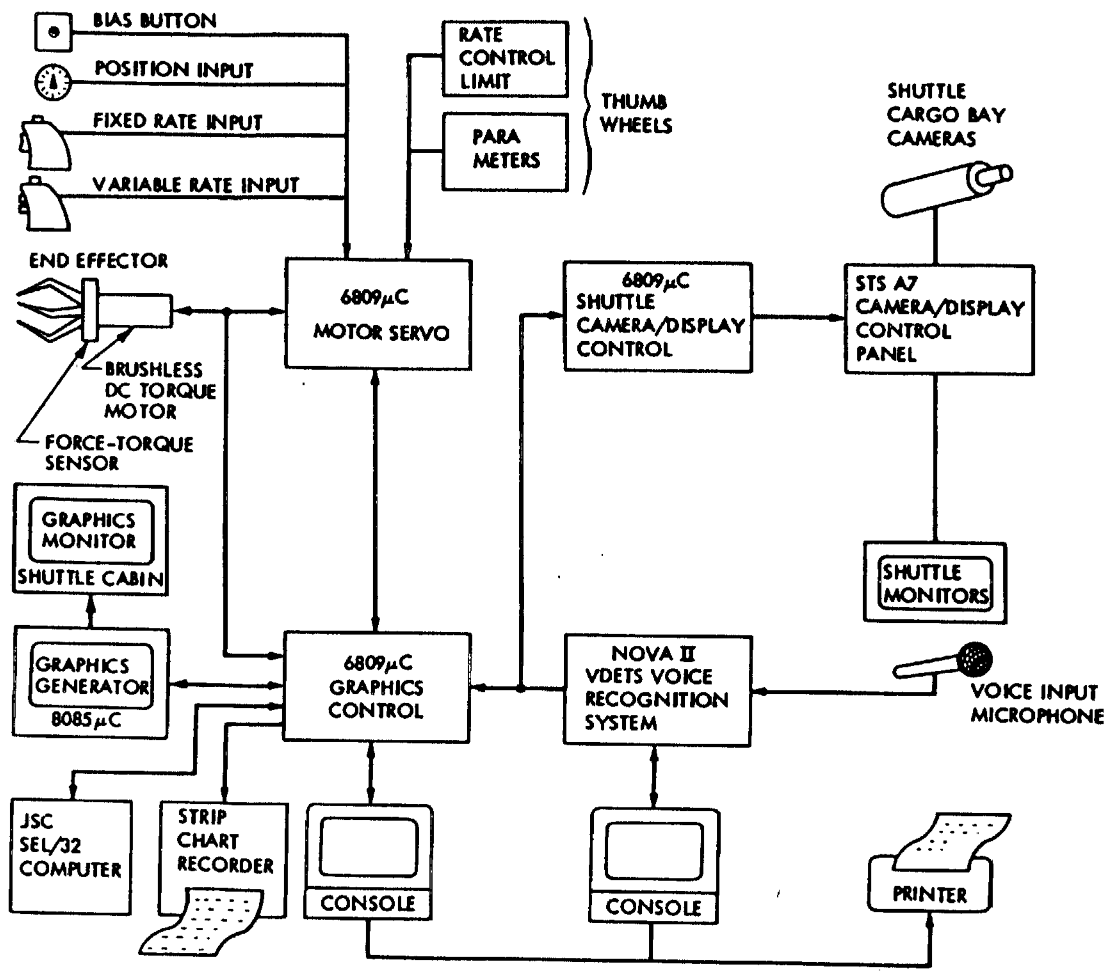
JPL R&D EXAMPLES IN CONTACT/NEAR CONTACT OPERATIONS

-- SENSOR REFERENCED CONTROL --

FORCE-TORQUE SENSOR





FORCE-TORQUE SENSOR WITH FOUR-CLAW END EFFECTOR

FOR SHUTTLE RMS SIZE ARM

The sensing concept and overall system configuration are shown here. More on this sensing and display system and on the ground experiments carried out at JSC using the full-scale simulated Shuttle RMS can be found:

1) A. K. Bejczy and R. S. Dotson, a Force-Torque Sensing and Display System for Large Robot Arms, Proceedings of IEEE Southeastcon'82, Destin, Fla., April 4-7, 1982.

2) A. K. Bejczy, R. S. Dotson, J. W. Brown and J. L. Lewis, Manual Control of Manipulator Forces and Torques Using Graphic Display, Proceedings of IEEE International Conference on Cybernetics and Society, Seattle, Wa. October 28-30, 1982.

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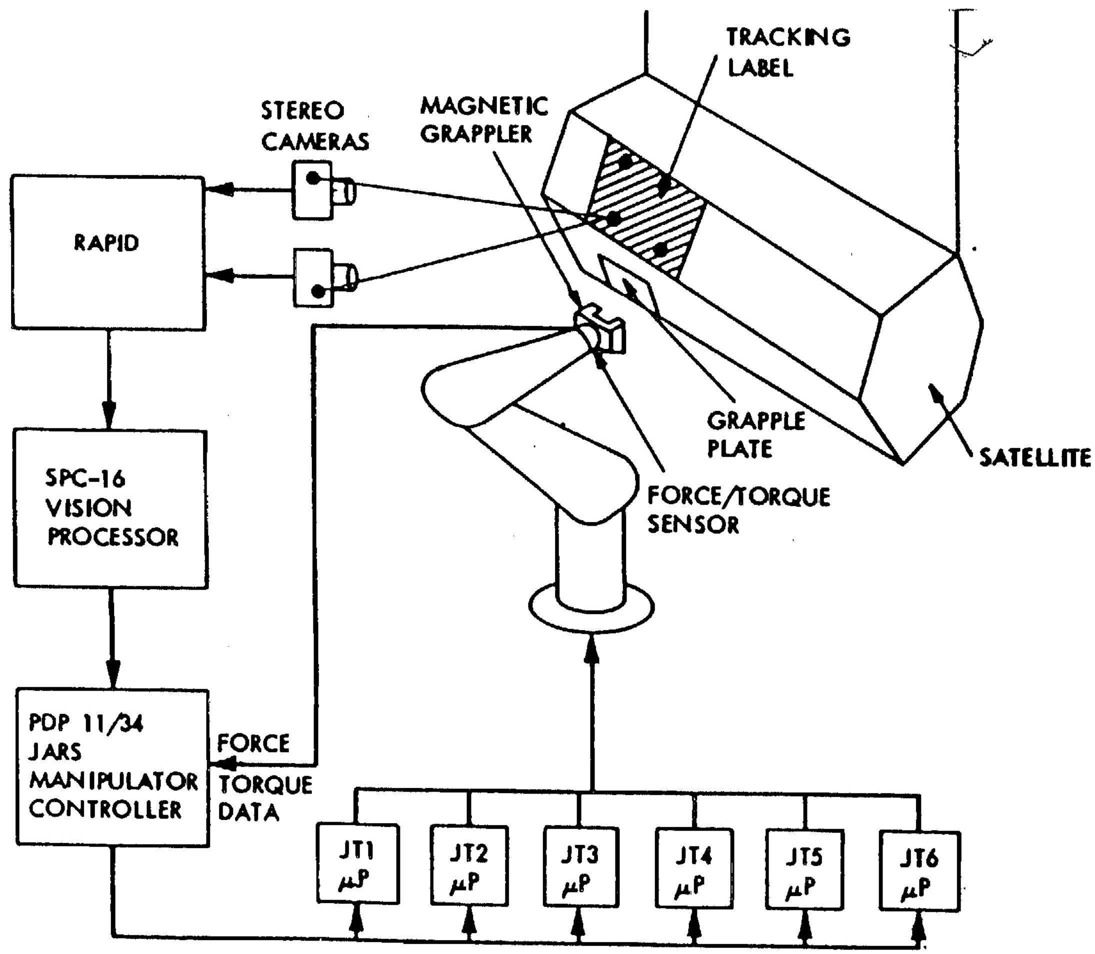
JPL R&D EXAMPLES IN CONTACT/NEAR CONTACT OPERATIONS

-- MACHINE VISION --

LABELED TRACKING FOR GRAPPLING



LABELED TRACKING FOR GRAPPLING

Real-time tracking and grappling of moving objects has been studied extensively at JPL in recent years. 30 Hz tracking of an object with visual labels has been demonstrated. The visual labels, three white dots on a dark background, were manually designated. Two solid-state cameras, spaced some 2 meters apart, allowed the system to triangulate on the centroid each of the dots, giving an accurate 3-D position and orientation of the simulated satellite. Grappling of the object was accomplished with a PUMA-600 manipulator arm with a compliant end-effector which limited the forces exerted by the arm on the satellite. The satellite, which started at over 1 m/s speed with respect to the arm, was brought to rest within 5 seconds.

Continuing work on this experiment includes modifying the PUMA control system to include force/torque feedback (as opposed to a passively-compliant end effector) to control the forces exerted by the arm. Also techniques for tracking of more general labels and automatic label acquisition are under study.

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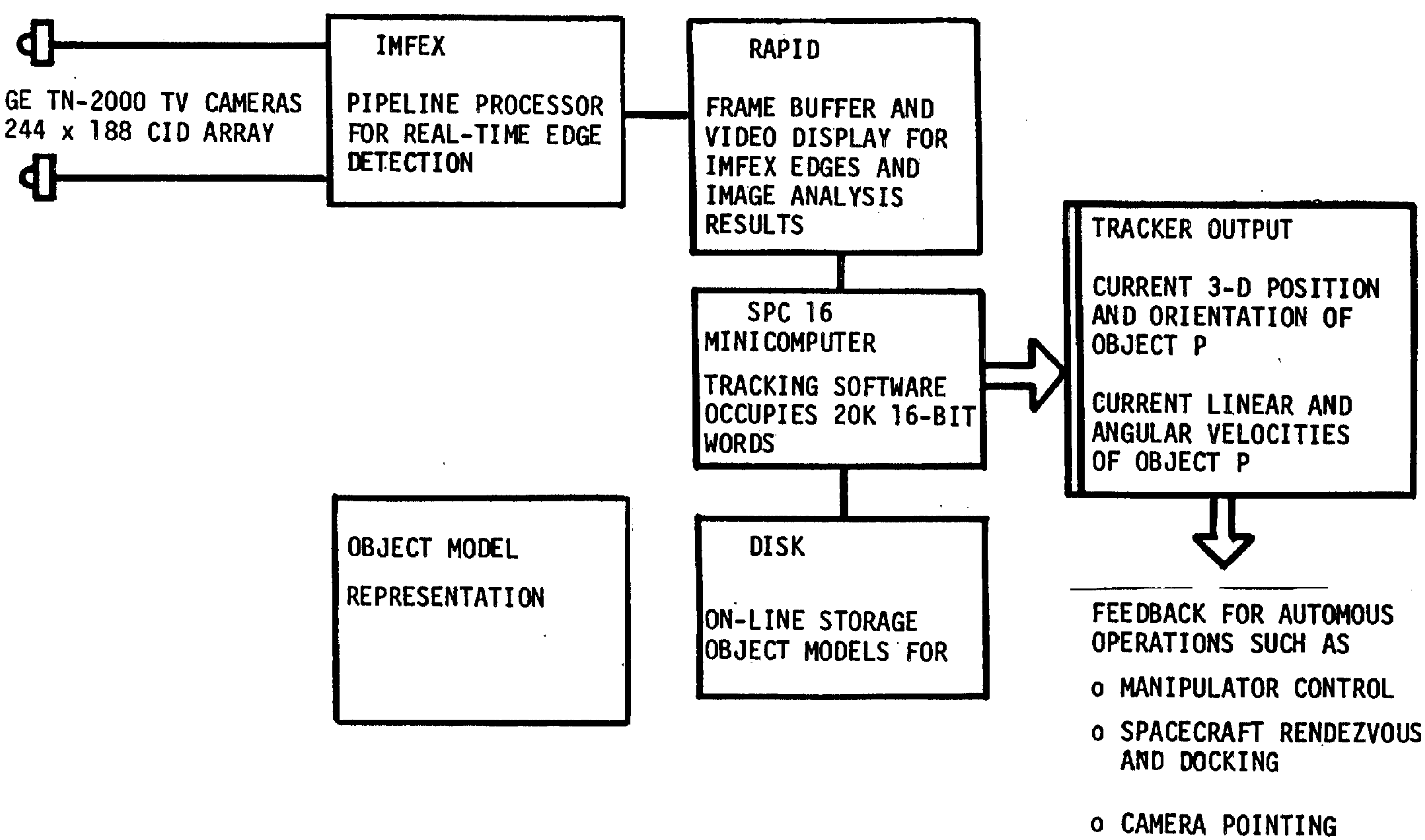
JPL R&D EXAMPLES IN CONTACT/NEAR CONTACT OPERATIONS

-- MACHINE VISION --

UNLABELED TRACKING FOR RECOGNITION AND MEASUREMENT



UNLABELED TRACKING FOR RECOGNITION AND MEASUREMENT

Unlabeled object tracking has also received considerable attention at JPL. A system capable of 2.5 Hz tracking of a convex polyhedra in spite of considerable background clutter and obscuration has been demonstrated. This work was based on a custom pipelined processor for edge enhancement and thinning which performs some 100 million operations per second. Current work includes automatic object acquisition, extending the object models to concave and curved objects, and interactive object model generation. An advanced pipelined processor, many times more capable than the existing one, is under development. When complete, this system will allow acquisition and 30 Hz tracking of complex unlabeled objects using wide-baseline imagery from any number of cameras.

5B-137

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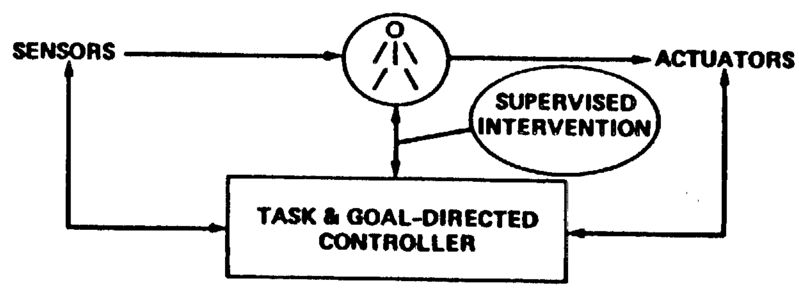
R&D CHALLENGES FOR ACHIEVING SYSTEM LEVEL SUPERVISED AUTONOMY

NEEDED → SYNERGISTIC INTEGRATION OF TELEROBOT SENSING, CONTROL, ACTUATION, AND THE OPERATOR

MISSING → PERCEPTUAL CAPABILITIES AND INTELLIGENT CONTROL FOR CONTEXT SENSITIVE TASK EXECUTION

REQUIRED → DEVELOPMENT OF TASK AND GOAL DRIVEN CONTROL



- KNOWLEDGE-BASED SENSING AND PERCEPTION
- TASK PLANNING
- CONFLICT RESOLUTION AND ERROR RECOVERY
- VERIFICATION AND EXECUTION MONITORING

58-138

R&D CHALLENGES

The challenges indicate that the achievement of increased system level supervised autonomy will require an intensive interdisciplinary activity. This activity will have to include crew training and acceptance.

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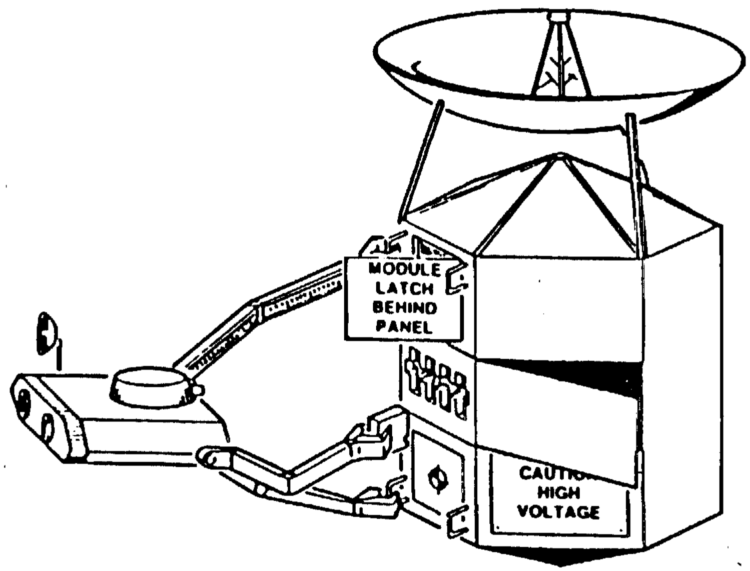
FIVE TECHNICAL THRUSTS TO ENABLE INTELLIGENT AUTOMATION



SENSING & PERCEPTION

NONCONTACT SENSOR UNDERSTANDING OF SCENE LAYOUT, SCENE MOTION, AND OBJECT IDENTITIES AND INTERRELATIONSHIPS. CONTACT SENSING FOR UNDERSTANDING OF OBJECT GEOMETRY, KINEMATICS AND DYNAMICS

TASK PLANNING & MANAGEMENT

LOGICAL REASONING AND SPATIAL PLANNING OF ALTERNATIVE TASK ACTIVITY SEQUENCES LEADING TO EFFICIENT TASK COMPLETION

INTELLIGENT CONTROL

PATTERN-DRIVEN RESPONSE TO SENSOR OBSERVATIONS. CHARACTERIZED BY COMBINING RULE-BASED ACTION WITH MODEL-BASED PARAMETER ADAPTATION

SUPERVISED ACTUATION

HUMAN FACTORS CONSIDERATIONS AND DISPLAY & CONTROL INTERFACE REQUIREMENTS ENABLING SMOOTH AND EFFICIENT TRANSITION OF CONTROL BETWEEN OPERATOR AND MACHINE

SYSTEM ARCHITECTURES & INTEGRATION

PROCESSOR ARCHITECTURES ENABLING ON-LINE EXECUTION OF INTELLIGENT AUTOMATION FUNCTIONS. SYSTEM CONTROL ARCHITECTURES ENABLING SMOOTH INTEGRATION OF, AND HUMAN INTERFACE WITH, THESE FUNCTIONS

27 March 1985

SESSION 6 - LASER AND RADIO FREQUENCY SYSTEMS AND TECHNOLOGY

6-1. "Review of Laser and RF Systems" - Harry Erwin and Kumar Krishen/NASA JSC

6-2. "Highly Stable Nd:YAG Laser Technology for Range and Range Rate Measurements" - R. Byer/Stanford University

6-3. "A Prospectus of Seminconductor and Advanced Solid State Laser Technology for Tracking/Rendezvous and Proximity Operations" - Peter Moulton and Richard Wangler/Schwartz Electro-Optics

6-4. "Long-Lifetime Stable CO_2 Lasers for Lidar and Comparison with Nd:YAG Lidar" - R. Hess, D. Schryer, B. Sidney, I. Miller, and G. Wood/NASA LaRC; B. Upchurch/Chemicon Inc; and K. Brown/Old Dominion University.

6-5. Review of Millimeter-Wave Component State-of-the-Art Technology" - Don Ball/Hughes

6-6. "A Millimeter Wave Range/Range Rate System for OMV Application" - E. Feagler/Allied Bendix Aerospace

6-7. "Rendezvous and Proximity Sensor Candidates" - B. Kunkel/MBB-ERNO

6-8. "Effects of Tethers on Rendezvous and Proximity Operations" - Joseph Carroll/California Space Institute

REVIEW OF LASER AND RF SYSTEMS

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JOHNSON SPACE CENTER
HOUSTON, TEXAS 77058

6-2

Johnson Space Center - Houston, Texas

Title	Division/Office	
REVIEW OF LASER AND RF SYSTEMS	TRACKING AND COMMUNICATIONS DIVISION	
	Presenter: ERWIN/KRISHEN	Date: FEB. 1985

PRESENTATION OUTLINE

- O PAST NASA SPACE PROXIMITY OPERATIONS
- O SHUTTLE KU-BAND SYSTEM
- O ANTICIPATED OPERATIONS/MISSIONS AND DATA REQUIREMENTS
- O FUTURE SYSTEMS/TECHNIQUES
    - O MICROWAVE
        - O GLOBAL POSITIONING SYSTEM
        - O MULTITARGET RADAR
        - O HIGHLY COMPACT RADAR
    - O LASER
        - O SYSTEM CONCEPTS
        - O IMPLEMENTATIONS
        - O TEST RESULTS
        - O TECHNOLOGY NEEDS

REVIEW OF LASER AND RF SYSTEMS

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Harry O. Erwin

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ABSTRACT

This paper presents a review of the ranging and tracking systems/techniques used in the past NASA programs. A review of the anticipated requirements for future rendezvous and docking operations is also presented as rationale for further development of the technology in this area. The first American rendezvous in space was between Gemini VI-A and Gemini VII and took place on December 15, 1965. The Gemini vehicles were equipped with a noncoherent pulse radar. The target vehicle carried a transponder to assist the radar in target acquisition. Angle tracking was accomplished by the phase-comparison monopulse technique. In the Gemini, Apollo, and Skylab programs, the rendezvous and/or docking were manual operations supported by radar measurements and visual observations. The Shuttle rendezvous radar is a Ku-band, pulse-Doppler radar which doubles as a communications transceiver. This radar is not accurate enough to support close-in stationkeeping or docking. An automatic soft-docking capability has been established as a requirement for future space operations. Millimeter wave and laser radar systems have shown promise in satisfying the needed accuracy requirements and size constraints (for space applications) compared to the microwave systems for proximity attitude, position, and velocity measurements. A review of these systems and their capabilities is presented in this paper. Rather than developing a separate sensor to satisfy the requirements of each new spacecraft, a hybrid design is proposed for a versatile system which can satisfy the needs for different spacecrafts and missions.

Title	Division/Office	
REVEIW OF LASER AND RF SYSTEMS	TRACKING AND COMMUNICATIONS DIVISION	
	Presenter: ERWIN/KRISHEN	Date: FEB. 1985

PAST SPACE PROXIMITY OPERATIONS

- O RENDEVOUS AND DOCKING (RAD) CAPABILITY DEVELOPED FIRST FOR GEMINI PROGRAM:
  - O FIRST RAD ACCOMPLISHED ON DECEMBER 15, 1965.
  - O L-BAND NONCOHERENT PULSE RADAR FOR RANGE, RANGE-RATE, ANGLE, ANGLE-RATE.
  - O ANGLE TRACKING BY PHASE-COMPARISON MONOPULSE TECHNIQUE.
  - O TARGET VEHICLE CARRIED TRANSPONDER TO ASSIST ACQUISITION.
  - O RAD MANUAL OPERATIONS AIDED BY VISUAL OBSERVATIONS.

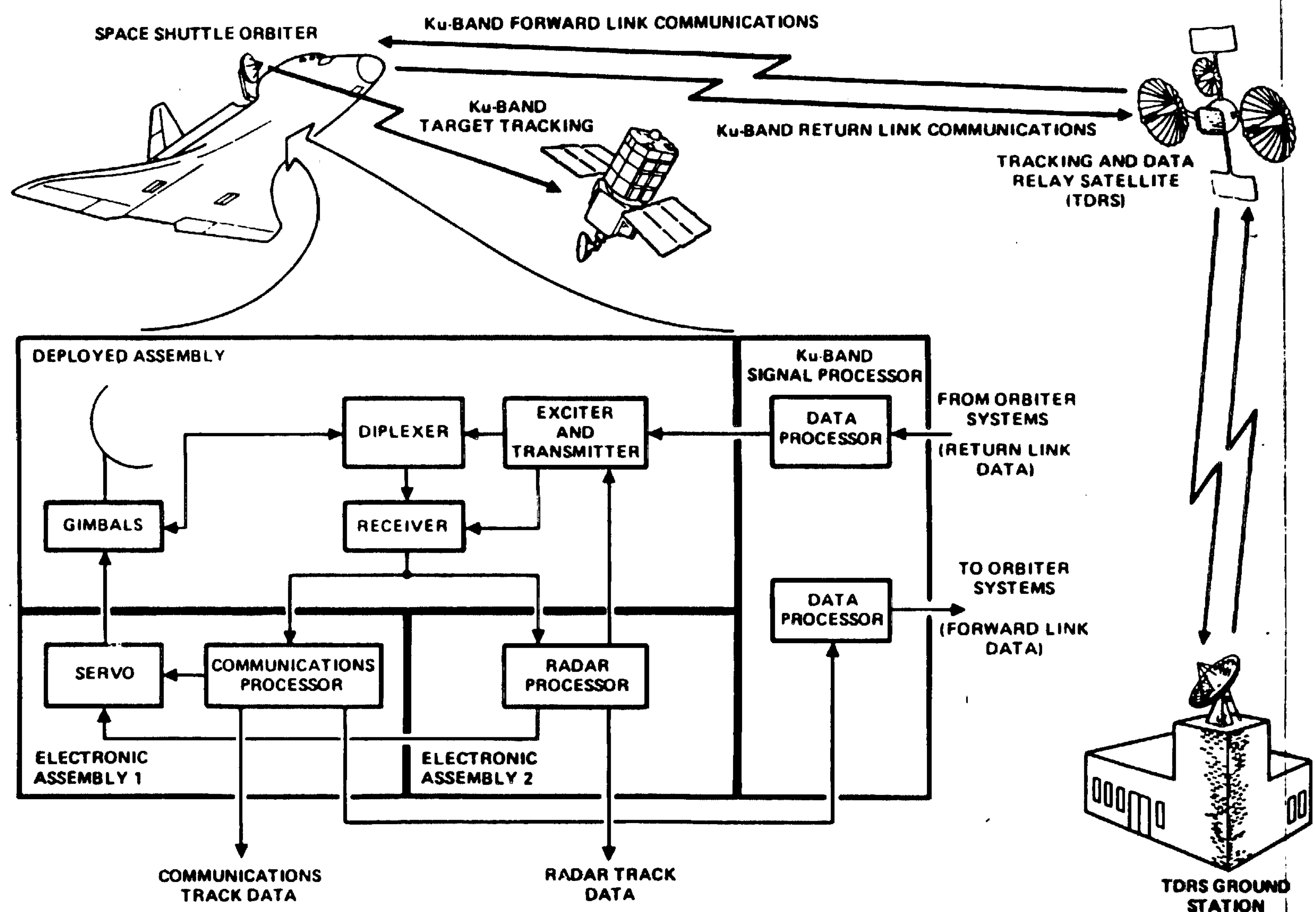
PAST SPACE PROXIMITY OPERATIONS (CONTINUED)

- O SEVERAL RAD OPERATIONS PERFORMED DURING APOLLO PROGRAM.
  - O LUNAR MODULE (LM) RAD WITH COMMAND AND SERVICE MODULE.
  - O LM X-BAND CW AMPLITUDE COMPARISON MONOPULSE.
  - O RANGE DETERMINED FROM PHASE SHIFTS ON THREE TONES (200Hz, 6.4KHz, AND 204.8KHz) THAT PHASE MODULATED THE CARRIER.
  - O ANGLES FROM AMPLITUDE-COMPARISON MONOPULSE TECHNIQUE.
  - O ANGLE-RATE FROM RADAR PEDESTAL RATE GYROS.
  - O PERFORMANCE PARAMETERS - ACCURACIES:
    - O RANGE: < 1% OVER 50 FEET MINIMUM AND SEVERAL HUNDRED MILES MAXIMUM RANGE.
    - O RANGE-RATE: < 1 FT/S.
    - O ANGLE: 2 MRAD.
    - O ANGLE-RATE: < 0.3 MRAD/S.
    - O RAD MANUAL OPERATIONS - RADAR MEASUREMENTS AND VISUAL OBSERVATIONS.
- O SKYLAB PROGRAM UTILIZED SAME TECHNIQUES AND SYSTEMS AS APOLLO FOR RAD OPERATIONS.

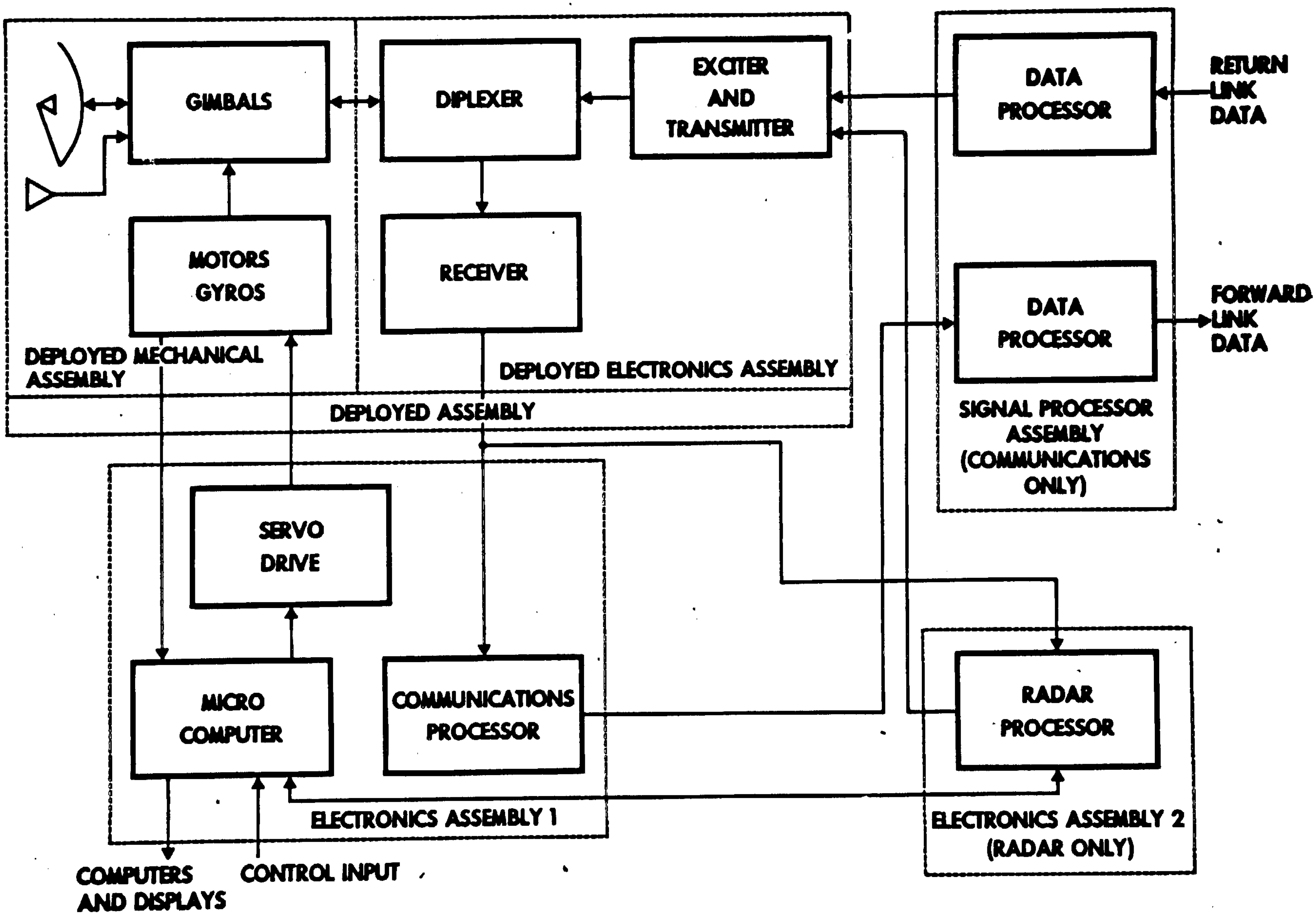
SHUTTLE RENDEZVOUS RADAR

- O KU-BAND, PULSE-DOPPLER RADAR WHICH DOUBLES AS COMMUNICATIONS TRANSCEIVER.
- O PARAMETERS DETERMINED AS FOLLOWS:
  - O RANGE FROM PULSE TRANSIT TIME.
  - O RANGE RATE FROM CARRIER DOPPLER SHIFT.
  - O ANGLE FROM AMPLITUDE-COMPARISON MONOPULSE TECHNIQUE.
  - O ANGLE-RATE FROM ANTENNA PEDESTAL GYROS.
- O PERFORMANCE PARAMETERS:
  - O MAXIMUM RANGE:
    - O 300NMI WITH TARGET TRANSPONDER
    - O 12NMI WITHOUT TARGET TRANSPONDER
  - O MINIMUM RANGE OF 100 FEET. SEARCH ACQ./TRACK VOLUME OF $\mp 30^{o}$.
  - O ERRORS (3 SIGMA) FOR $1M^2$ TARGET CROSS-SECTION:
    - O RANGE: GREATER OF 80 FEET OR 1%
    - O RANGE-RATE: 1 FT/S
    - O ANGLE: 8 MRAD
    - O ANGLE-RATE: 0.14 MRAD/S
- O NOT INTENDED FOR CLOSE-IN STATION-KEEPING OR DOCKING.

ORBITER Ku-BAND RADAR/COMMUNICATIONS SYSTEM



INTEGRATED Ku BAND FUNCTIONAL DIAGRAM



KU-BAND SYSTEM TESTS ON THE ORBITER

FUTURE ANTICIPATED USES AND DATA ACCURACIES FOR RAD SENSORS

SPACE ACTIVITIES

	SPACE SATELLITES	SPACE STATION	SHUTTLE/ VEHICLES	UNKNOWN OBJECTS
DEPLOYMENT	X	X	X	
CONSTRUCTION		X	X	
OPERATION	X	X	X	
INSPECTION	X	X	X	X
REPAIR	X	X	X	
RETRIEVAL	X	X	X	X

SPACE MISSIONS

	RENDEZVOUS	STATIONKEEPING	DOCKING
DEPLOYMENT		X	
CONSTRUCTION	X	X	X
OPERATION	X	X	X
INSPECTION	X	X	
REPAIR	X	X	X
RETRIEVAL	X	X	X

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REVIEW OF LASER AND RF SYSTEMS	TRACKING AND COMMUNICATIONS DIVISION	
	Presenter: ERWIN/KRISHEN	Date: FEB. 1985

FUTURE ANTICIPATED USES AND DATA ACCURACIES FOR RAD SENSORS (CONCLUDED)

ANTICIPATED RENDEZVOUS AND DOCKING DATA REQUIREMENTS/ACCURACIES

PARAMETER	LIMITS	ACCURACY (3 SIGMA)
RANGE	1M-200 KM	0.01 X RANGE
RANGE-RATE	± 30M/S	0.01M/S (R ≤ 1M/S)
		0.1 M/S (R > 1M/S)
ANGLE	± 0.25 RAD	10 MRAD
ANGLE-RATE	± 10 MRAD/S	0.1 MRAD/S
ATTITUDE (P,Y)	± 0.25 RAD (R < 100 M)	10 MRAD
ATTITUDE (R)	± π RAD (R < 100 M)	30 MRAD
ATTITUDE-RATE	± 20 MRAD/S (R < 100 M)	0.1 MRAD/S

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	Presenter: ERWIN/KRISHEN	Date: FEB. 1985

SENSOR CHARACTERISTICS

- O SMALL SIZE/WEIGHT AND LOW COST.
- O LONG LIFE AND LOW COST OF MAINTENANCE.
- O SENSOR TO ALLEVIATE/REDUCE OPERATIONAL CONSTRAINTS.
- O MODULAR DESIGN TO CONFIGURE FOR VARIETY OF MISSIONS, VEHICLES, ORBIT CONDITIONS, VEHICLE APPROACHES, AND PERFORMANCE PARAMETERS.

GLOBAL POSITIONING SYSTEM

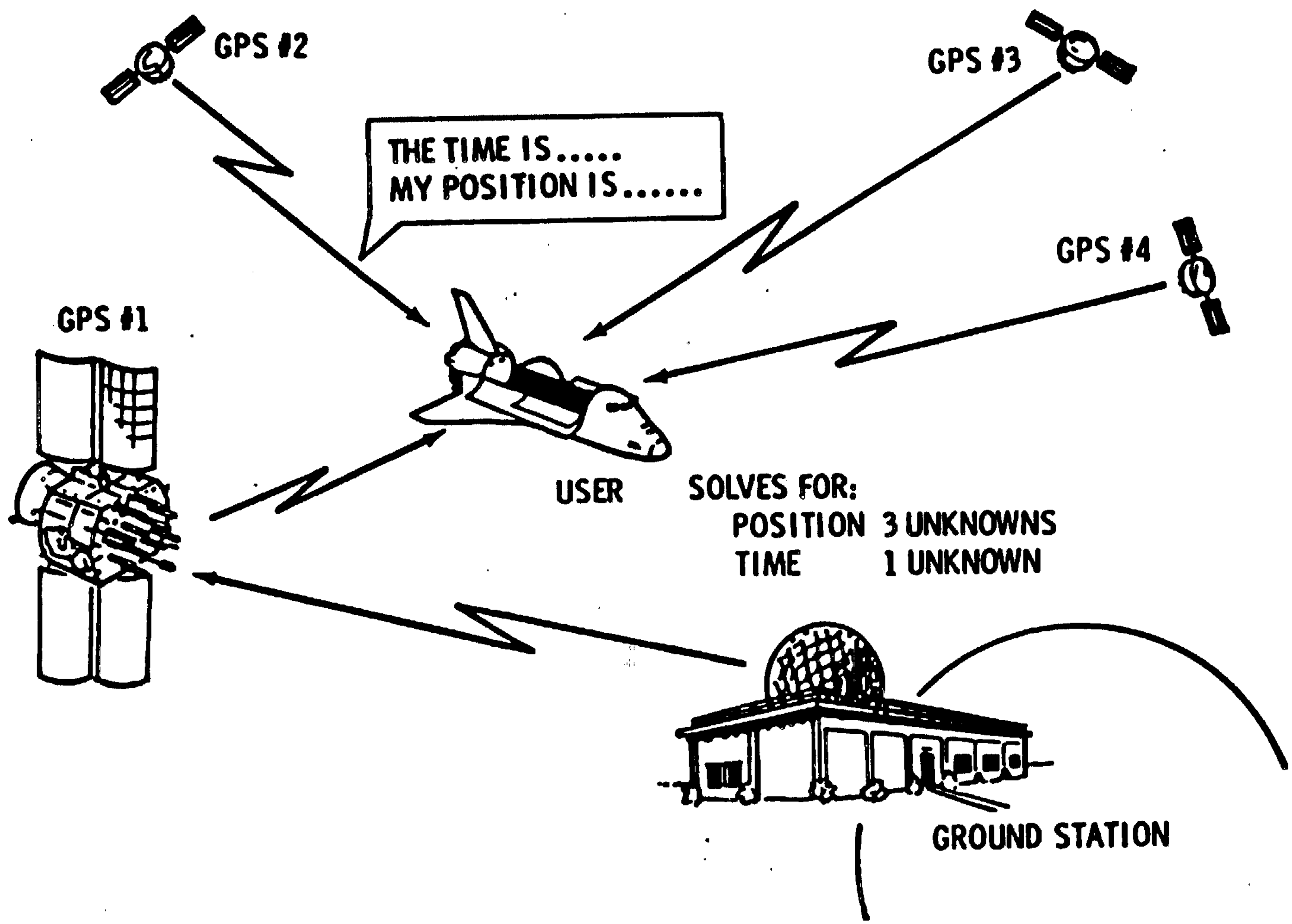
- DOD NAVSTAR GLOBAL POSITIONING SYSTEM (GPS) IS RADIO NAVIGATION SYSTEM WHICH PROVIDES MEASUREMENT OF TIME, VELOCITY, AND 3D POSITION.
- EVENTUALLY GPS WILL CONSIST OF 18 SPACE VEHICLES (SV'S) IN 6 ORBITAL PLANES WITH 3 SATELLITES PER PLANE.
- SIGNAL CHARACTERISTICS:
  - SPACE VEHICLE TRANSMITS 2 CARRIER SIGNALS IN L-BAND.
  - TWO PSEUDO-RANDOM NOISE CODES ON ONE CARRIER AND ONE SUCH CODE ON THE OTHER CARRIER.
  - NAVIGATION DATA MODULATED ON CARRIERS AT 50 BITS PER SECOND.

GLOBAL POSITIONING SYSTEM (CONCLUDED)

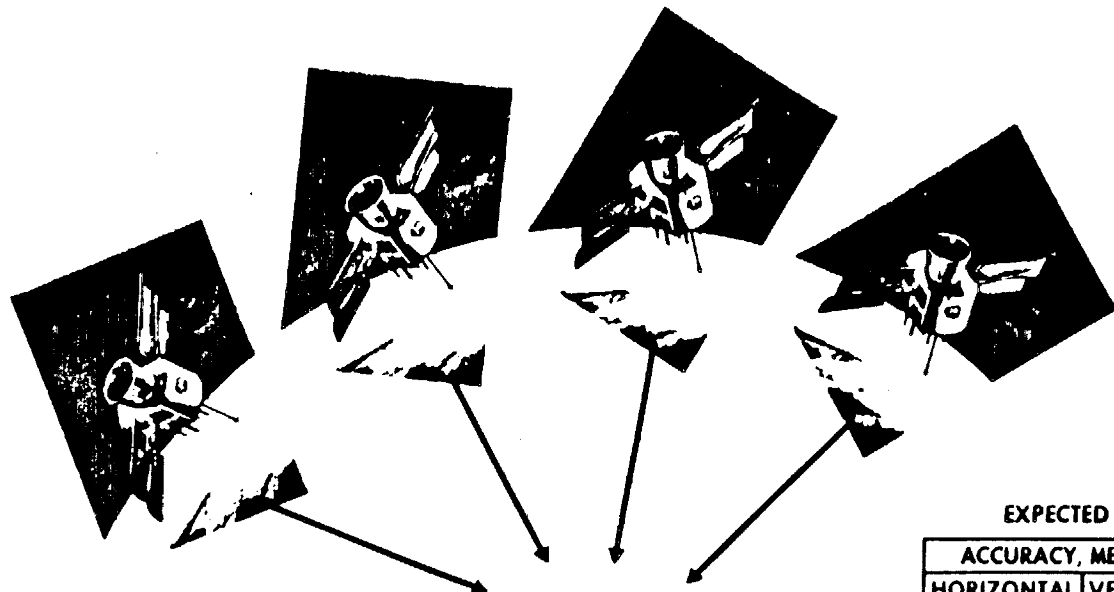
O TECHNIQUE:

- O TRANSIT TIME FROM SV DETERMINED FROM PHASE DIFFERENCE IN CODE SEQUENCE BETWEEN SIGNAL TRANSMITTED AND RECEIVED TIMES.
- O RANGE FROM 4 SV'S DETERMINED USING TRANSIT TIME.
- O USER RECEIVER/PROCESSOR ASSEMBLY (RPA) TO AUTOMATICALLY SELECT 4 VISIBLE SV'S WITH BEST GEOMETRY.
- O EXPECTED ACCURACIES/COVERAGE:
  - O POSITION COORDINATES: 10M-RMS
  - O POSITION RANGE: RATE 0.3MPS-RMS
  - O COVERAGE: CONTINUOUS

NAVIGATING WITH GPS



UTILIZATION OF GLOBAL POSITIONING SYSTEM BY SHUTTLE ORBITER



FOUR INPUTS FOR
POSITION DETERMINATION

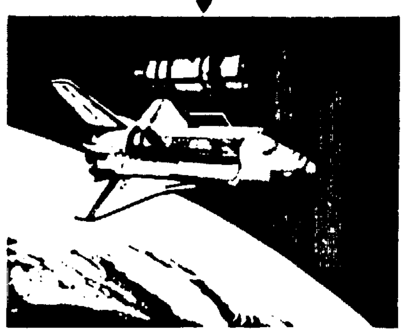
EXPECTED GPS PERFORMANCE

ACCURACY, METERS		PERCENT OF TIME THAT ACCURATE
HORIZONTAL	VERTICAL	
5	7	50
9	10	90

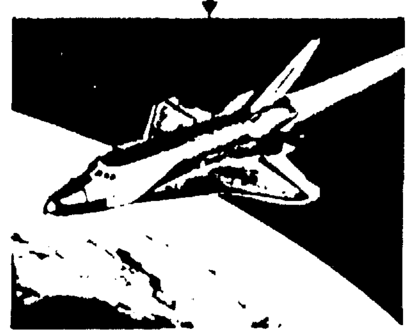
FREQUENCIES:

L_1 1676.42 MHz

L_2 1227.6 MHz



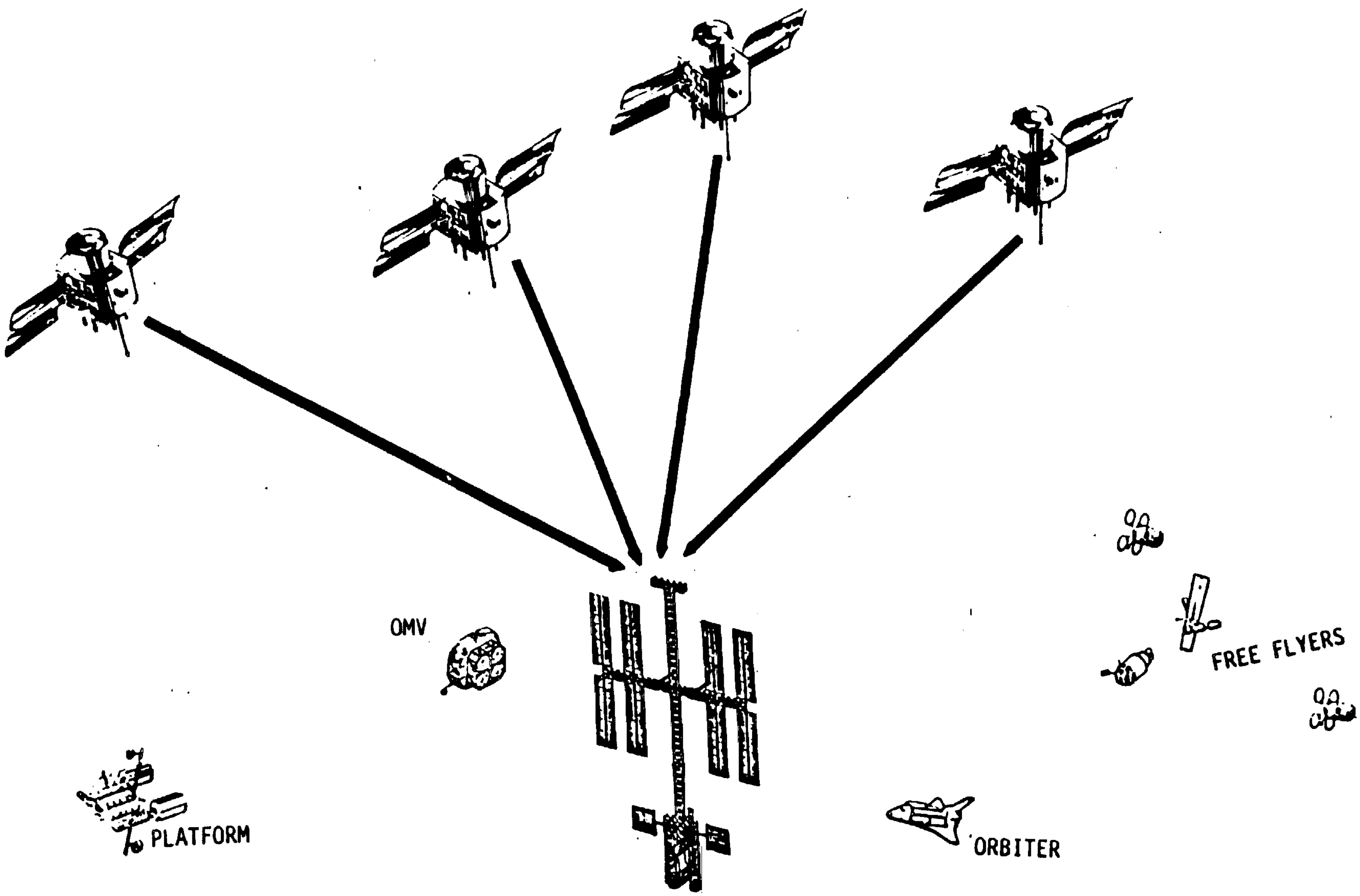
S.IUTTLE POSITION DURING
ORBITAL OPERATIONS



SHUTTLE POSITION PRIOR
TO DE-ORBIT

SHUTTLE POSITION AFTER BLACKOUT

GLOBAL POSITIONING SYSTEM



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MULTI-TARGET TRACKING MICROWAVE SYSTEM

- O SIGNIFICANT PARAMETERS:
  - O X-BAND, PULSE DOPPLER
  - O PLANAR, PHASED ARRAY ANTENNA
  - O SOLID STATE TRANSMITTER
- O ELECTRONIC BEAM STEERING:
  - O SAVES POWER/WEIGHT BY ELIMINATING GIMBAL STRUCTURES
  - O PROVIDES POTENTIAL FOR HIGHER ANGLE ACCURACY
  - O INSTANTANEOUS ACQUISITION OF MULTIPLE TARGETS
- O SOLID STATE TRANSMITTER IMPROVES PERFORMANCE:
  - O INCREASED RELIABILITY
  - O NO HIGH VOLTAGE
  - O NO VIBRATION CONCERNS
- O FAULT TOLERANT ARCHITECTURE IN ANTENNA/TRANSMITTER EXHIBITS GRACEFUL DEGRADATION.

MULTI-TARGET TRACKING MICROWAVE SYSTEM (CONCLUDED)

- O EXPECTED PERFORMANCE PARAMETERS:
  - O ACCURACIES:
    - O RANGE: ± 10 FT 3 SIGMA
    - O RANGE-RATE: ± .2 FT/S
    - O ANGLE: ± .01 RAD
    - O ANGLE-RATE: 0.10 MRAD/SEC.
  - O COVERAGE:
    - O RANGE: 100 FT TO 20 NM
    - O RANGE-RATE: 32 FT/S MAX.
    - O SPATIAL: ± 30^{o} AZ AND EL

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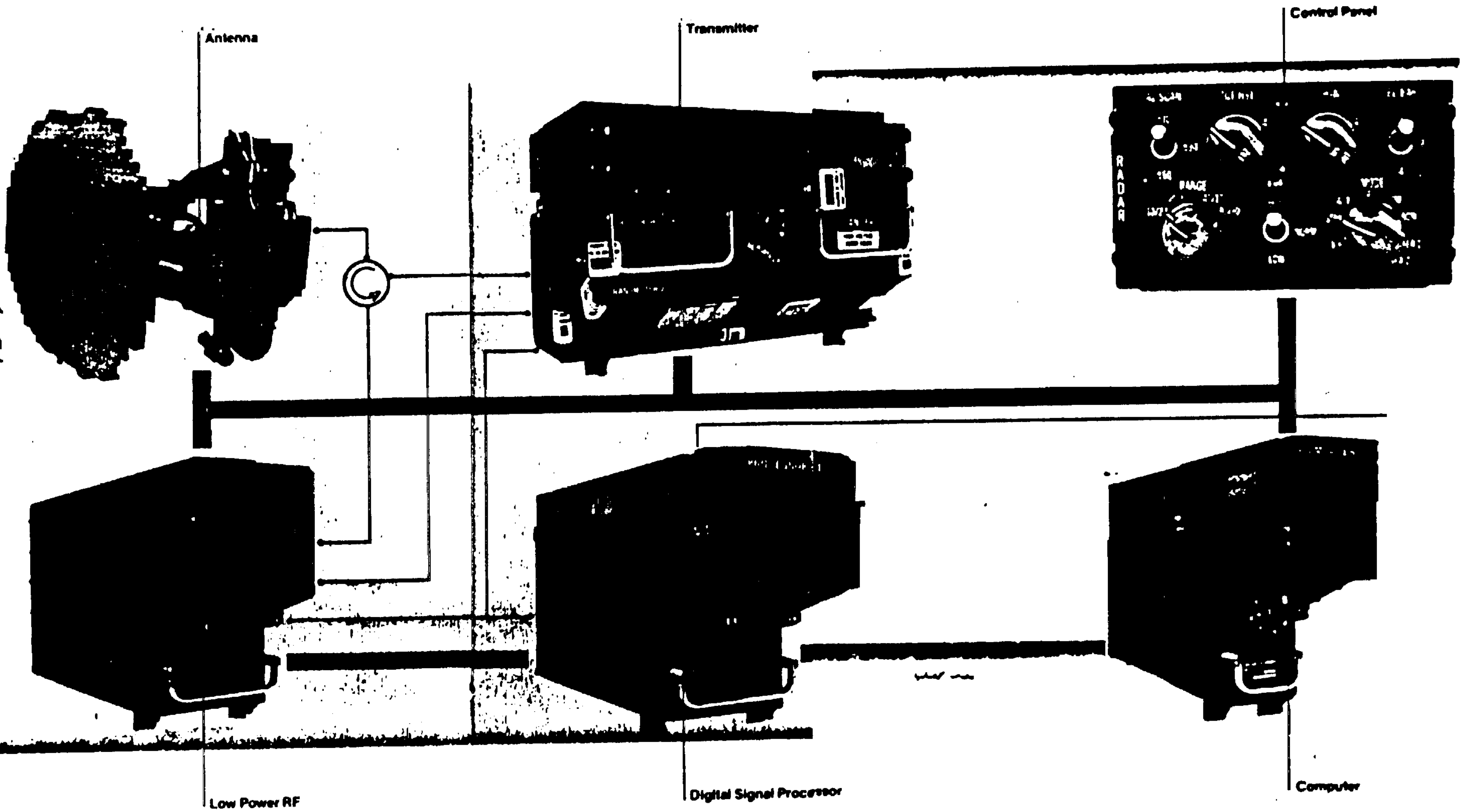

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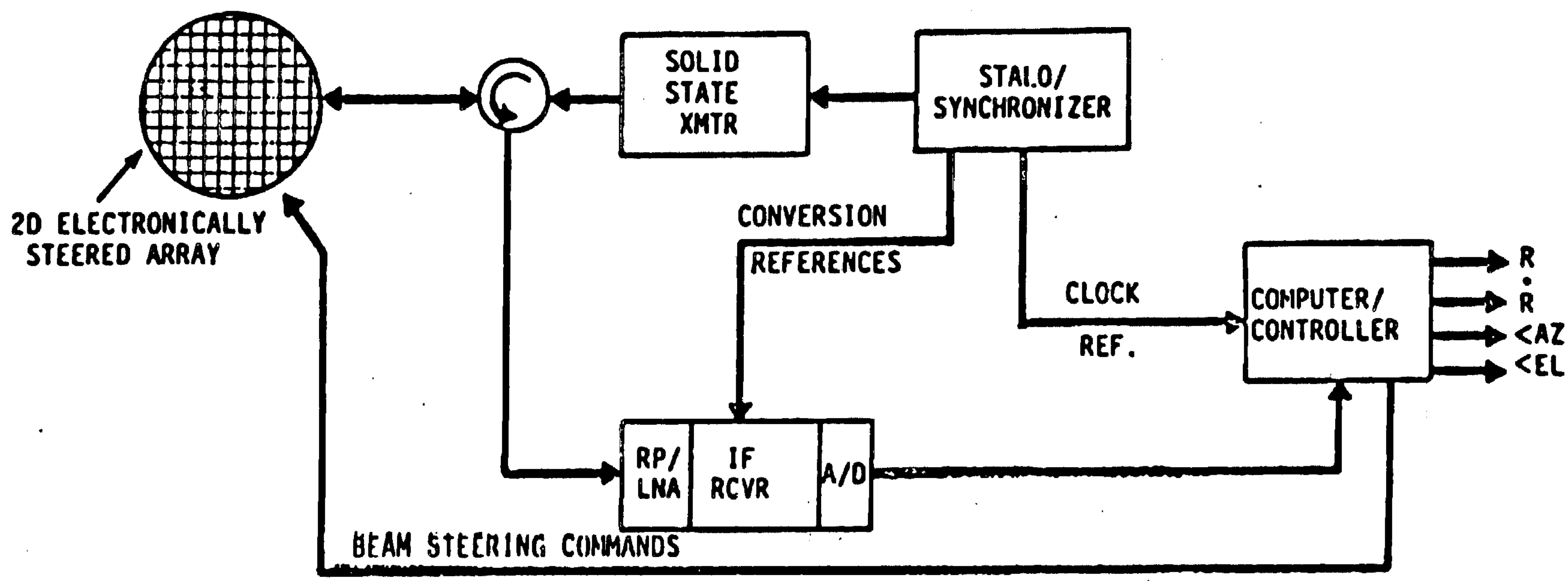
	Presenter	Date
	ERWIN/KRISHEN	FEB. 1985



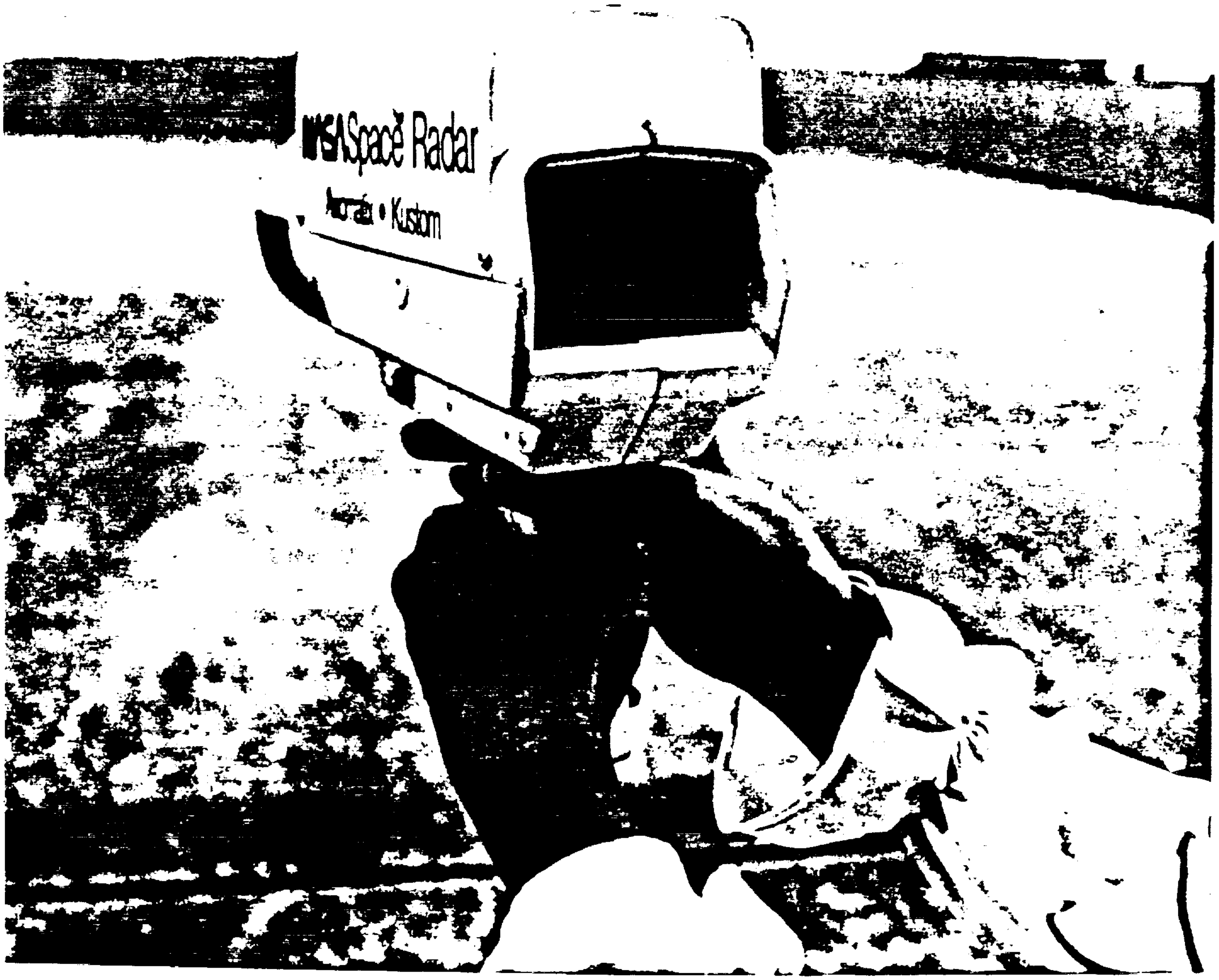
Shuttle Experiment Radar

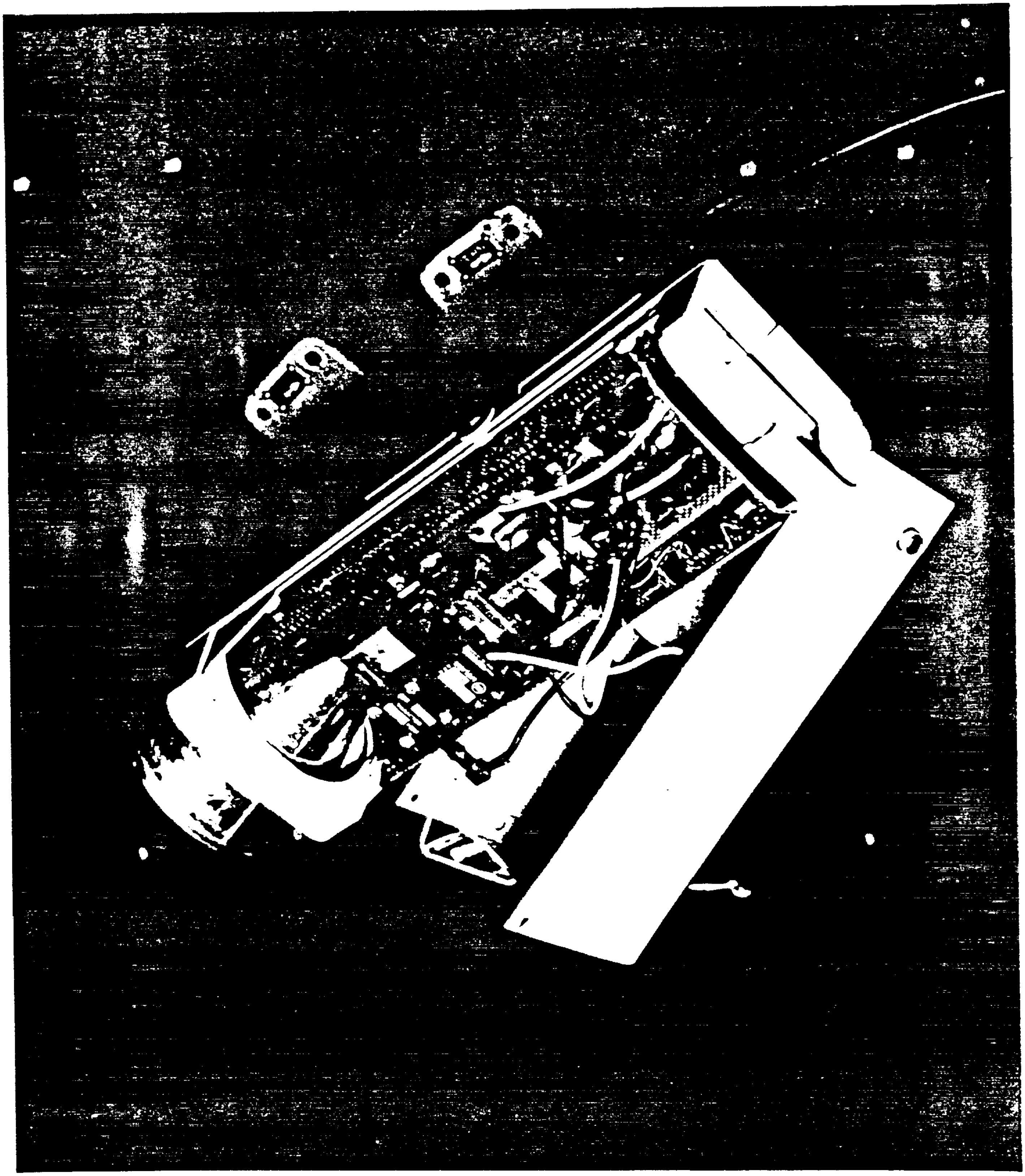
PROXIMITY OPERATIONS PORTABLE RADAR

(IN-HOUSE DEVELOPMENT)

SIZE	APPROXIMATELY 10 X 12 X 3.5" (SAME APPROXIMATE SIZE AS THE HAND-HELD POLICE SPEED RADAR)
WEIGHT	4 LBS
POWER REQUIREMENTS	12V (NORMALLY SUPPLIED BY BATTERY PACK), 1.5 AMPS
FREQUENCY OF OPERATION	24 GHz (NOMINAL)
ANTENNA BEAMWIDTH	9^{0} (3 dB)
ANTENNA GAIN	26 dB
MAXIMUM DESIGN RANGE	6000 FT
DESIGN GOAL RANGE ACCURACY	0.33 M (SIGMA)
DESIGN GOAL VELOCITY ACCURACY	0.01 M/S (SIGMA)
RF POWER OUT	190 MW

PROXIMITY OPERATIONS PORTABLE RADAR





PROXIMITY OPERATIONS PORTABLE RADAR

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FUTURE SYSTEMS	TRACKING AND COMMUNICATIONS DIVISION	
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MM WAVE HIGHLY COMPACT RADAR RANGING

- O SMALL, LIGHTWEIGHT RADAR MOUNTED ON MMU FOR EVA ACTIVITY.
- O SPACE SYSTEM DESIGN TO INCORPORATE:
  - O SIGNIFICANT REDUCTION IN POWER CONSUMPTION.
  - O HIGHER ACCURACIES NEAR ZERO-RANGE RATE REGION.
  - O EASE OF OPERATION AND UNIQUE DATA DISPLAY.
  - O HIGH RELIABILITY
- O SYSTEM PARAMETERS/GOALS:
  - O OPERATING FREQUENCY: 100 GHz.
  - O ACCURACIES:
    - O RANGE: 5 FEET
    - O RANGE-RATE: .05 FT/S
    - O ANGLE: .01 RAD
    - O ANGLE-RATE: 0.10 MRAD/S
  - O COVERAGE:
    - O RANGE: 0 TO 500 FEET
    - O RANGE-RATE: 10 FPS MAX
    - O SPATIAL: $\pm 10^{\circ}$ AZ AND EL

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Title	Division/Office	
HIGHLY COMPACT RADAR RANGING	TRACKING AND COMMUNICATIONS DIVISION	
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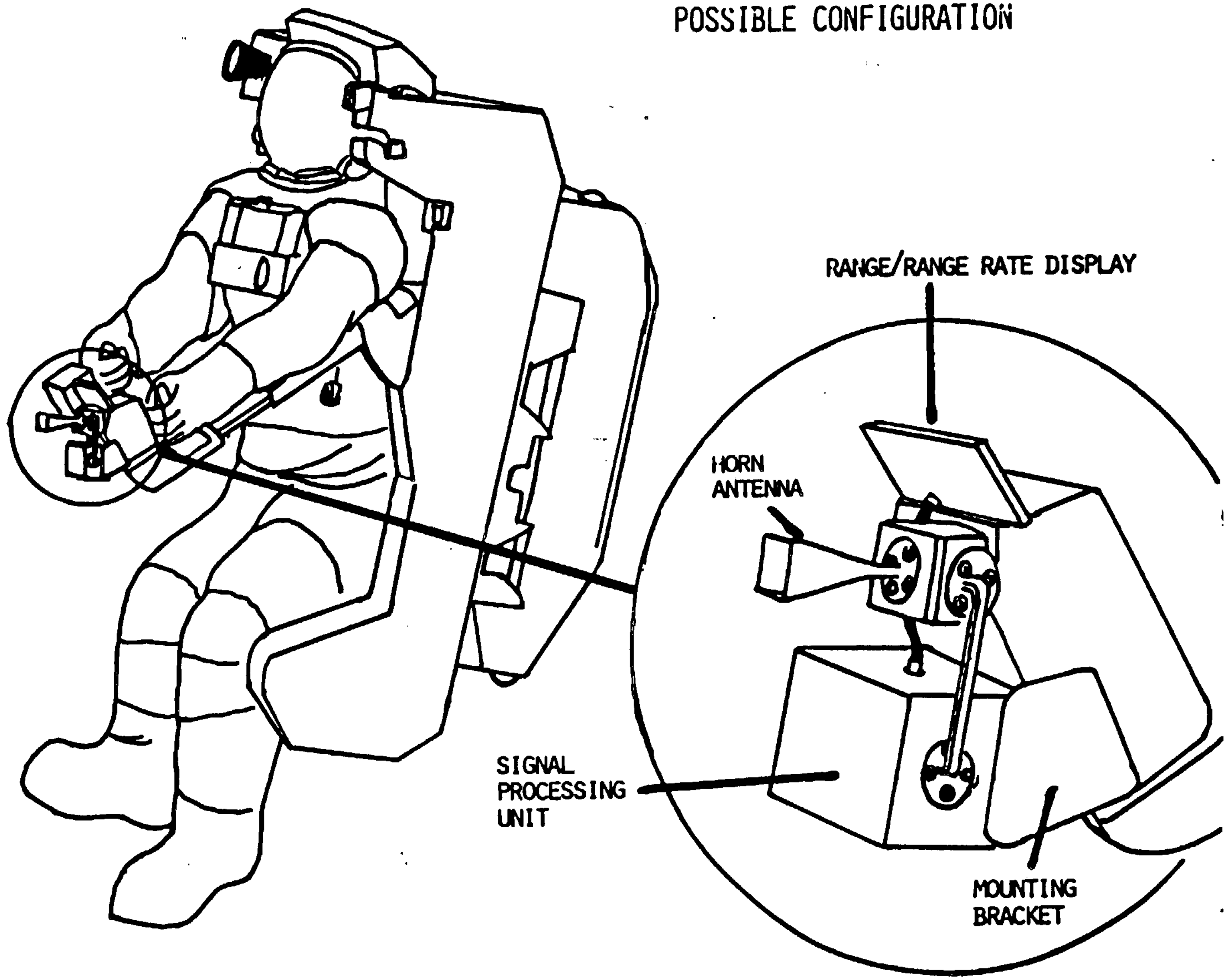
POSSIBLE CONFIGURATION



LASER RANGING/TRACKING/DOCKING SYSTEM

0 POTENTIAL ADVANTAGES

- 0 SMALL SIZE/WEIGHT.
- 0 BETTER ACCURACY AT NEAR-RANGE AND LOW VELOCITIES.
- 0 HIGH RELIABILITY USING SOLID STATE LASERS.
- 0 CAN PROVIDE TARGET ATTITUDE INFORMATION.
- 0 BEAM STEERING AND MULTIMODE IMPLEMENTATIONS CAN INCREASE ANGULAR COVERAGE AND ALLEVIATE OPERATIONAL CONSTRAINTS.

0 IMPLEMENTATION CONFIGURATIONS

- 0 PULSED RADAR.
- 0 CHIRPED RADAR.
- 0 TONE RANGING RADAR.
- 0 ONE WAY RADAR.

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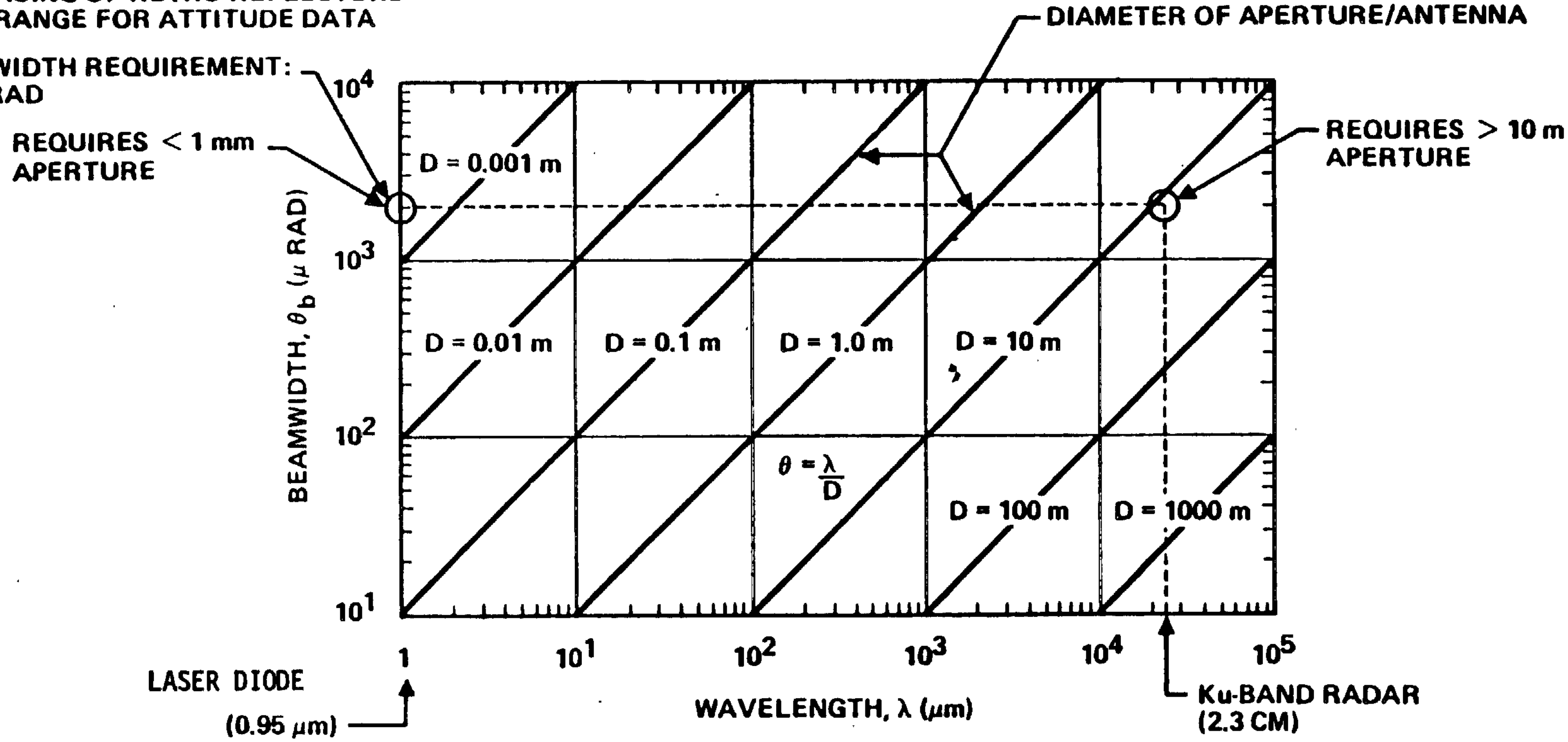
Title	Division/Office	
REVIEW OF LASER AND RF SYSTEMS	TRACKING AND COMMUNICATIONS DIVISION	
	Presenter: ERWIN/KRISHEN	Date: FEB. 1985

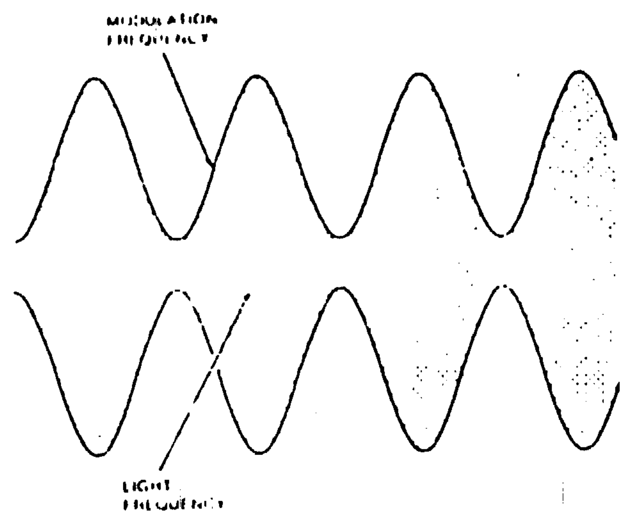
COMPARISON OF LASER AND Ku-BAND APERTURE REQUIREMENTS

FOR:
1 m SPACING OF RETRO-REFLECTORS
100 m RANGE FOR ATTITUDE DATA

BEAMWIDTH REQUIREMENT: ≤2 m RAD



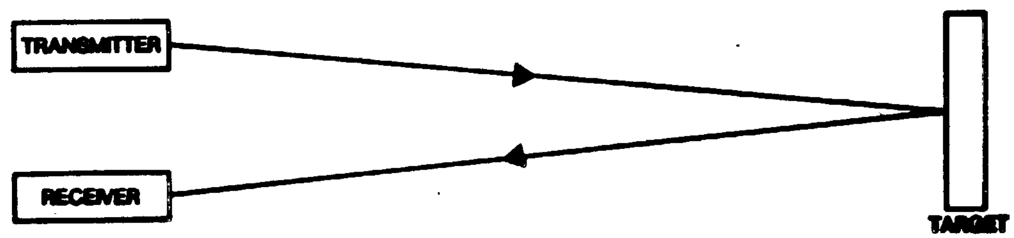
BEAMWIDTH VS WAVELENGTH FOR A UNIFORMLY ILLUMINATED CIRCULAR APERTURE



Modulated light wave

LASER DOPPLER

DOPPLER EFFECT = FREQUENCY CHANGE DUE TO MOTION



FREQUENCY TRANSMITTED = f_T

FREQUENCY REFLECTED = $f_T \pm f_D$

$f_D = \frac{2V}{\lambda}$

V = VELOCITY
λ = WAVELENGTH

MEASURING f_D PROVIDES A MEASURE OF V

EXAMPLE:

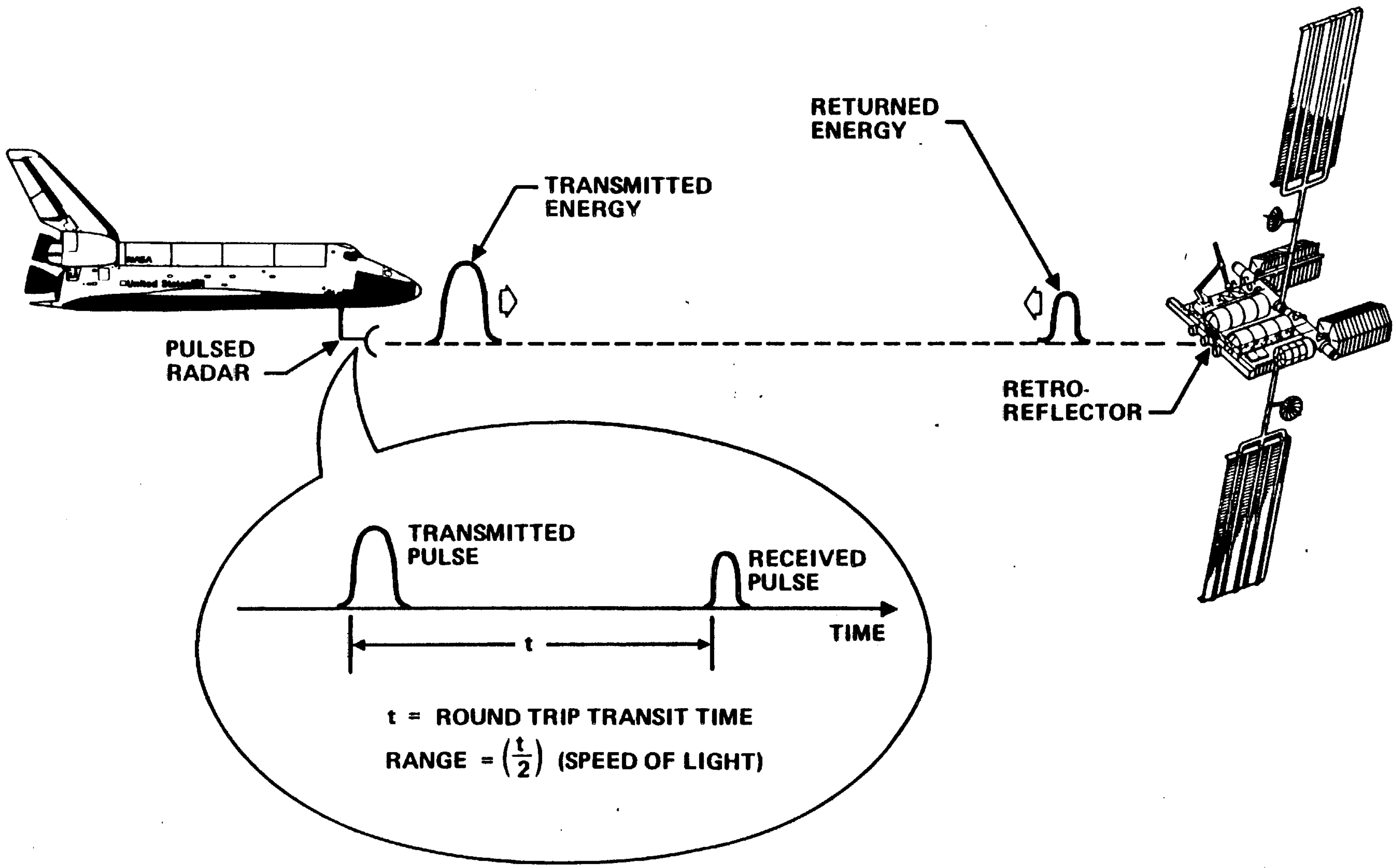
f_D = 30 KHZ

λ = $.6328 \times 10^{-6}$ M

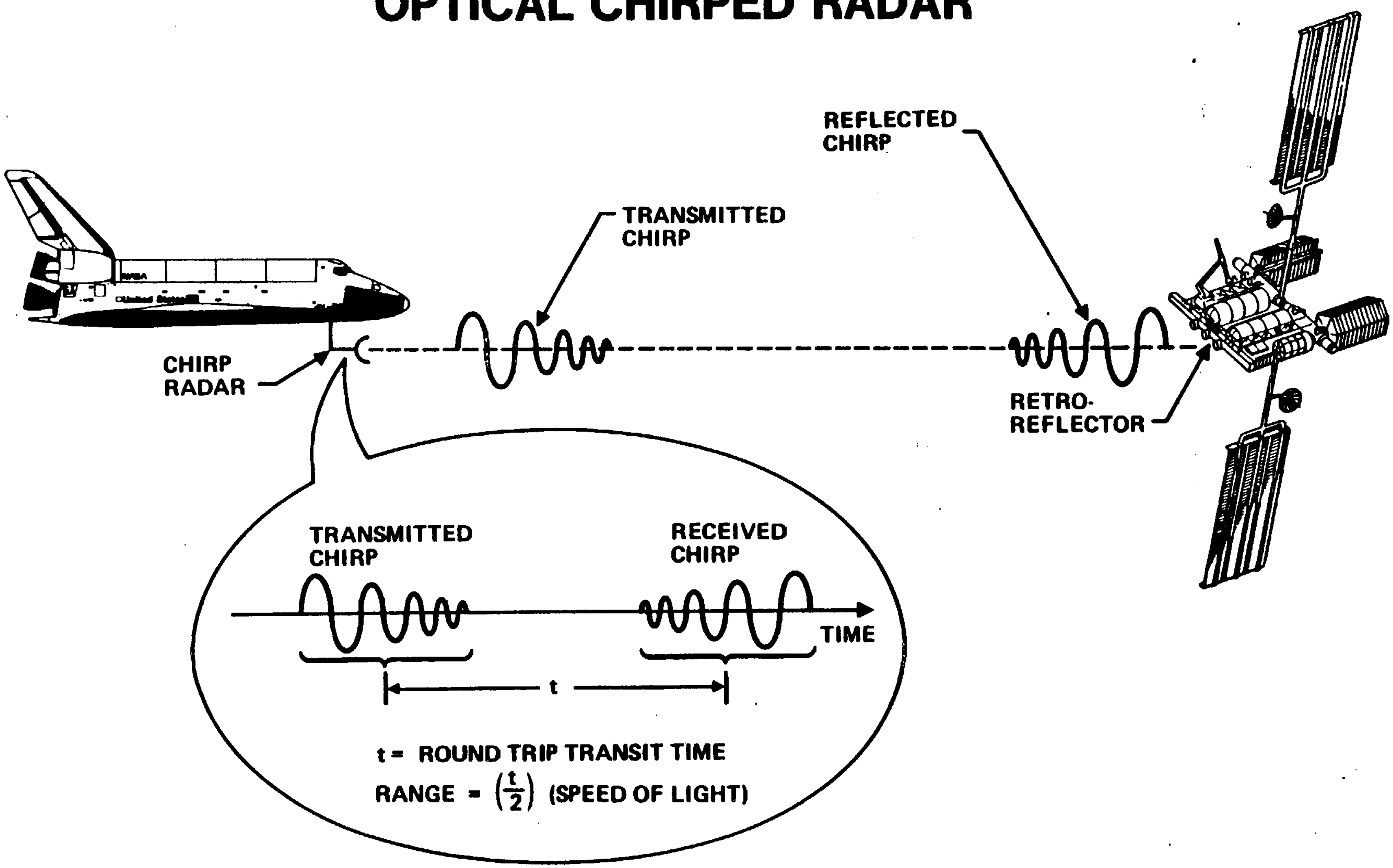
$V = \frac{f_D \lambda}{2} = \frac{30 \times 10^3 \times .6328 \times 10^{-6}}{2}$

$= \frac{.949 \text{ CM}}{\text{SEC}}$

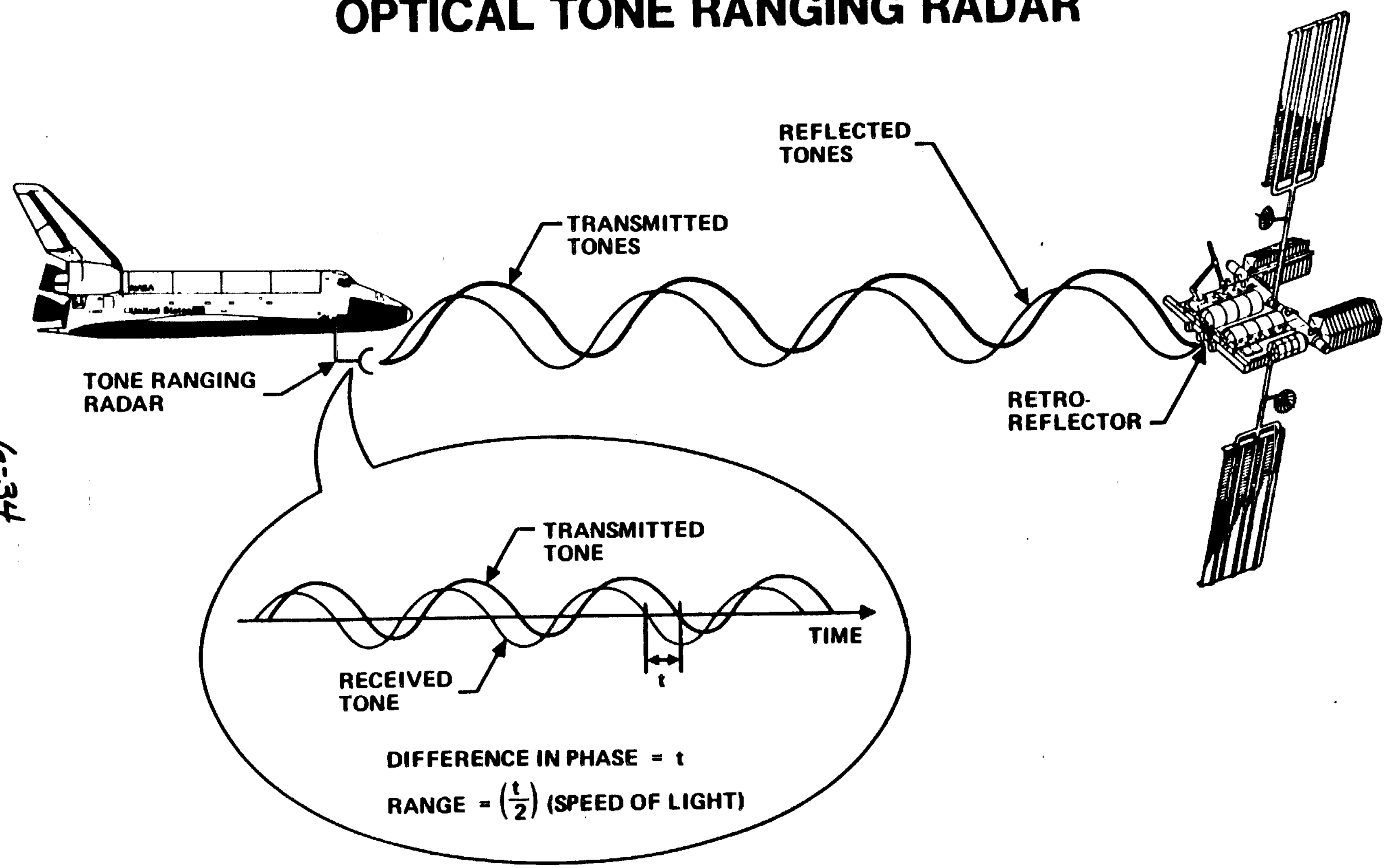
OPTICAL PULSED RADAR



OPTICAL CHIRPED RADAR



OPTICAL TONE RANGING RADAR



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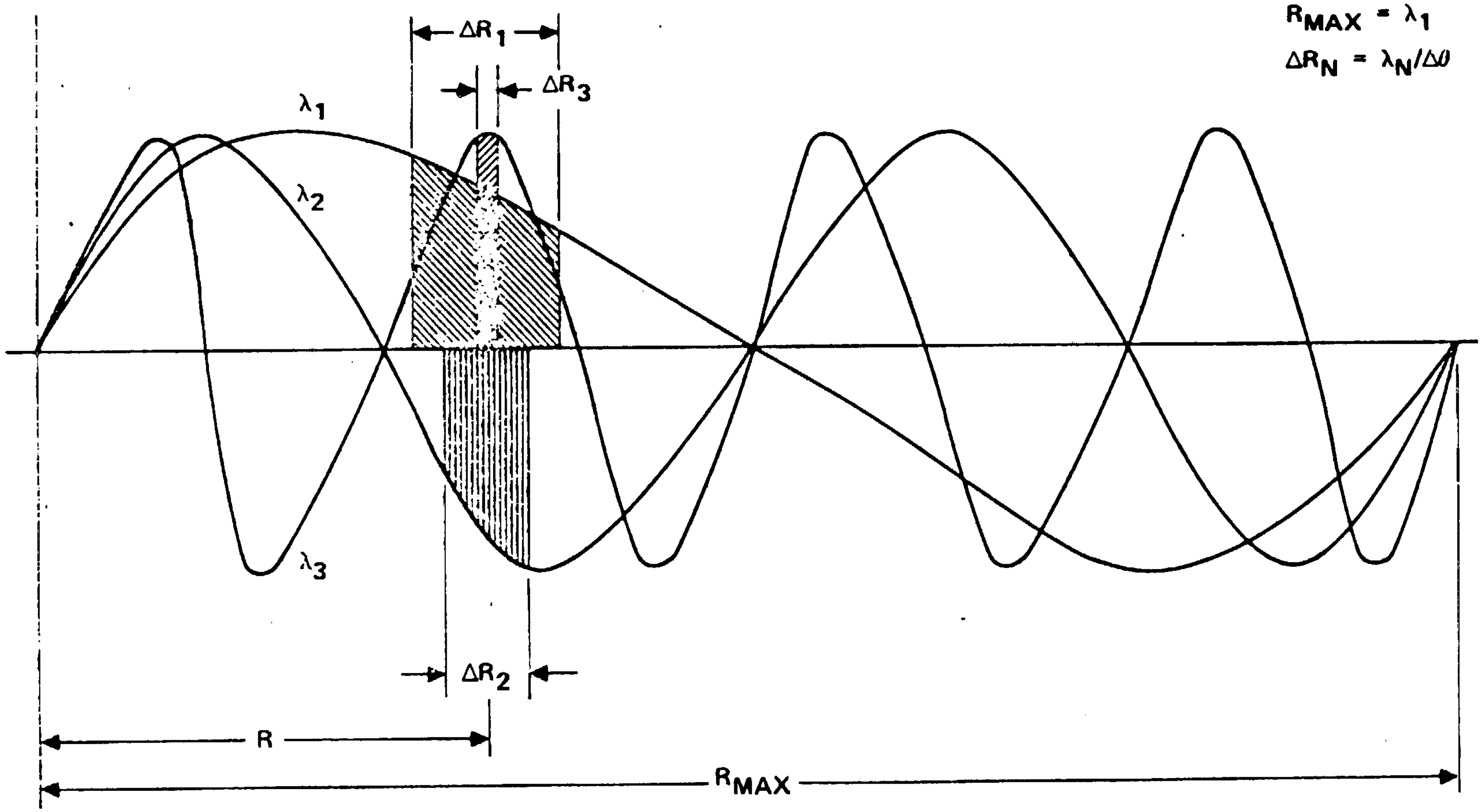
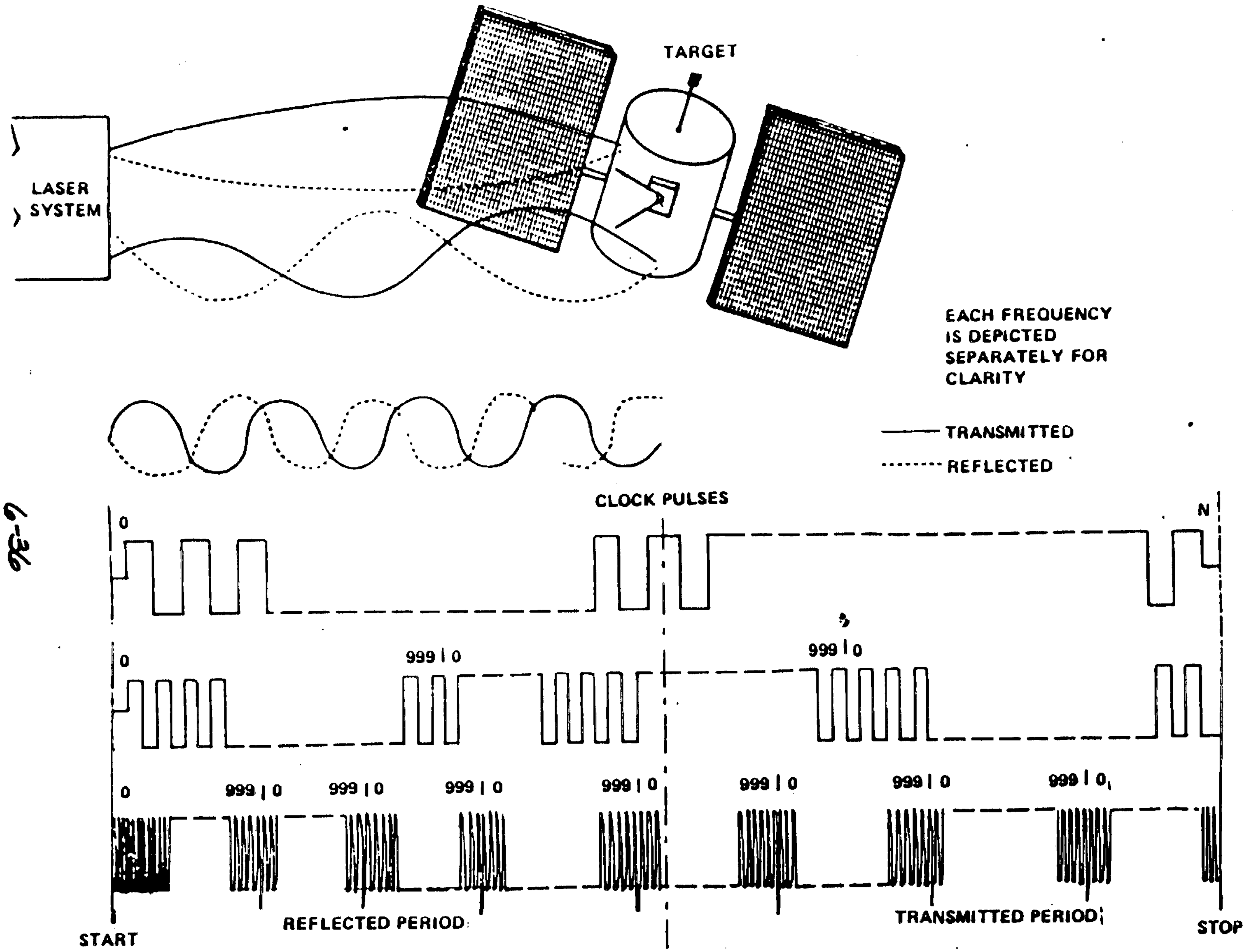
TONE RANGING

$R_{MAX} = \lambda_1$

$\Delta R_N = \lambda_N / \Delta\theta$





Multiple frequency tone ranging.

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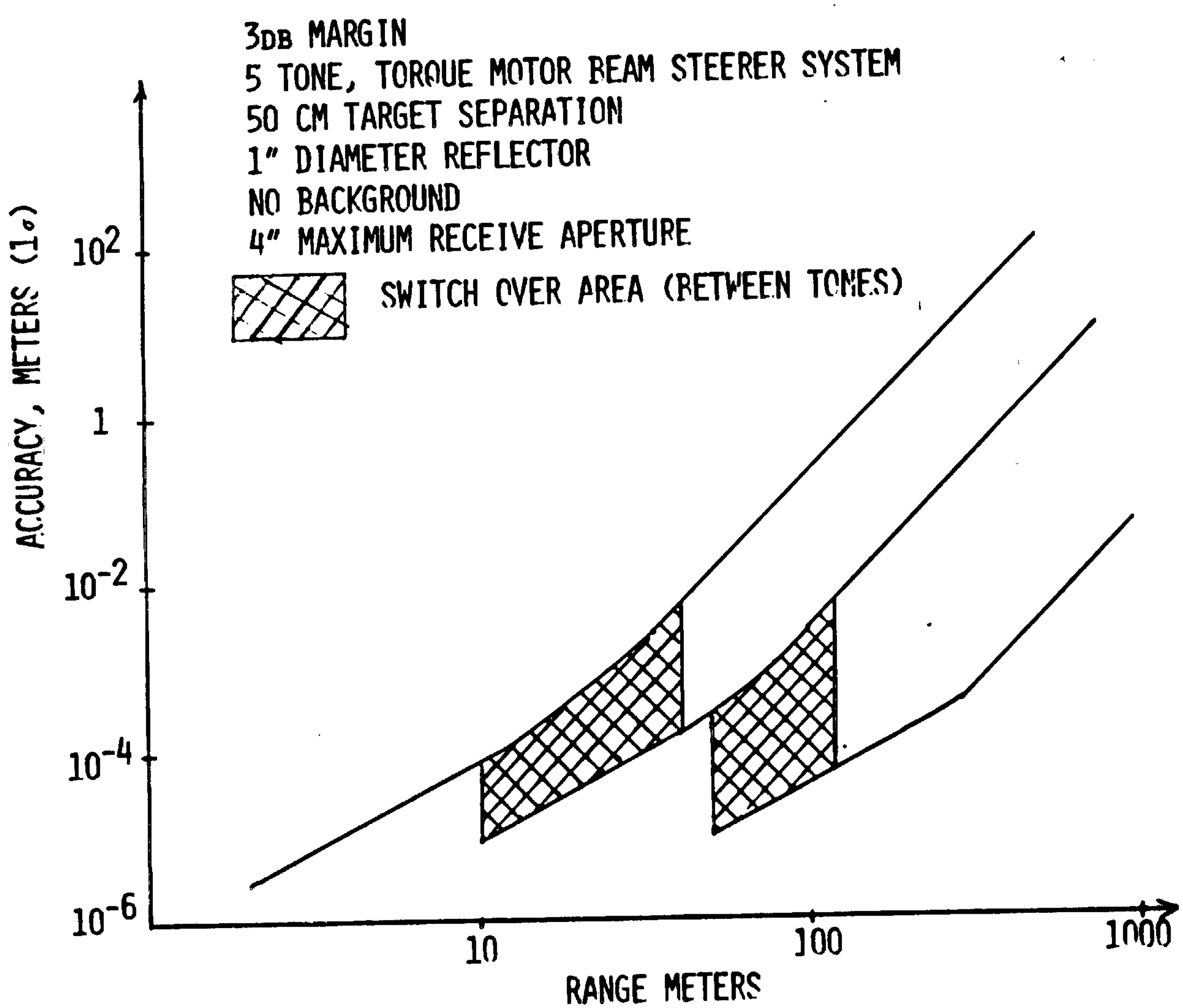
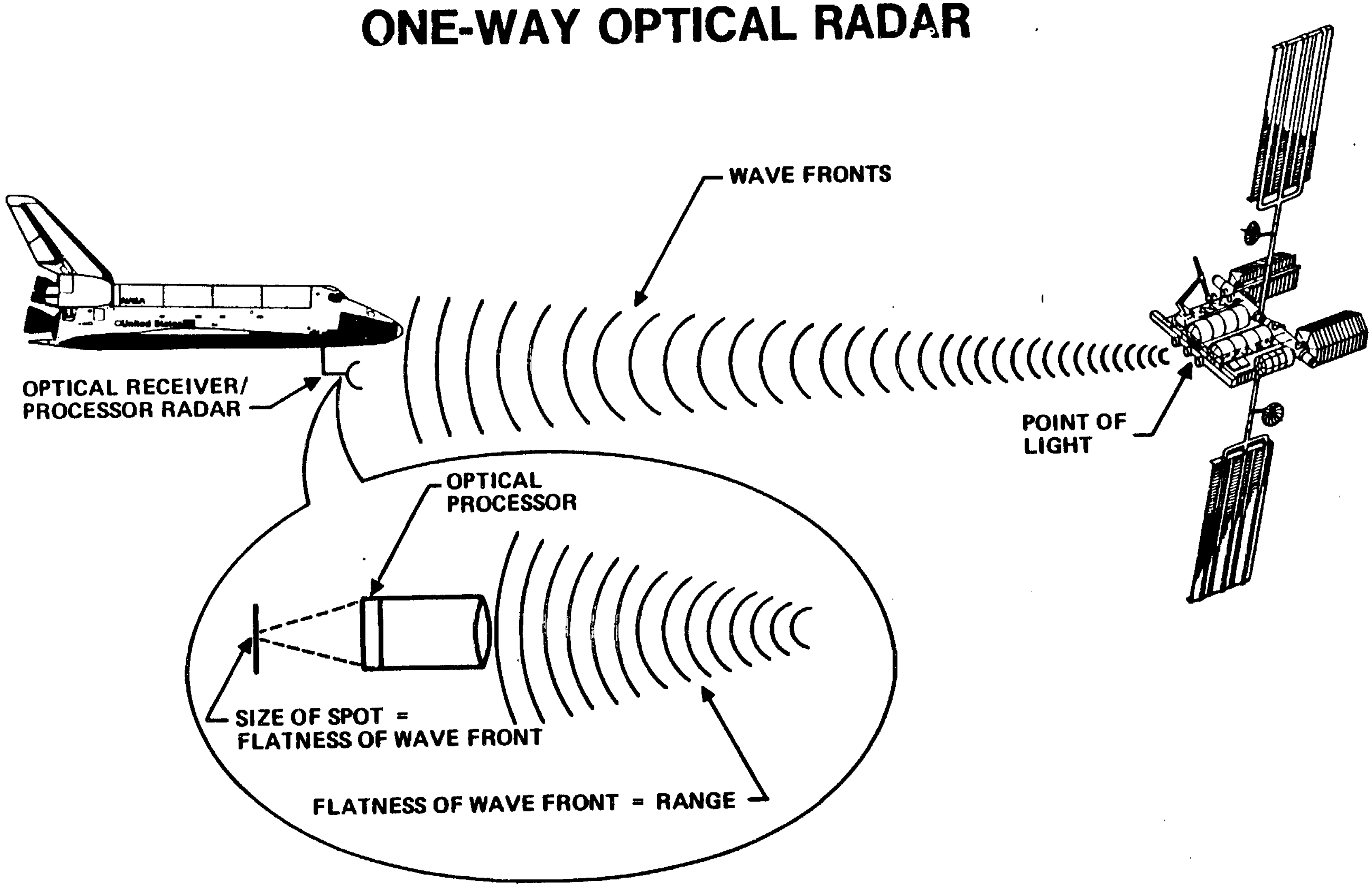
Title	Division/Office	
LASER DOCKING SYSTEMS TESTS	TRACKING AND COMMUNCIATIONS DIVISION	
	Presenter: ERWIN/KRISHEN	Date: FEB. 85

ACCURACY VS RANGE



ONE-WAY OPTICAL RADAR



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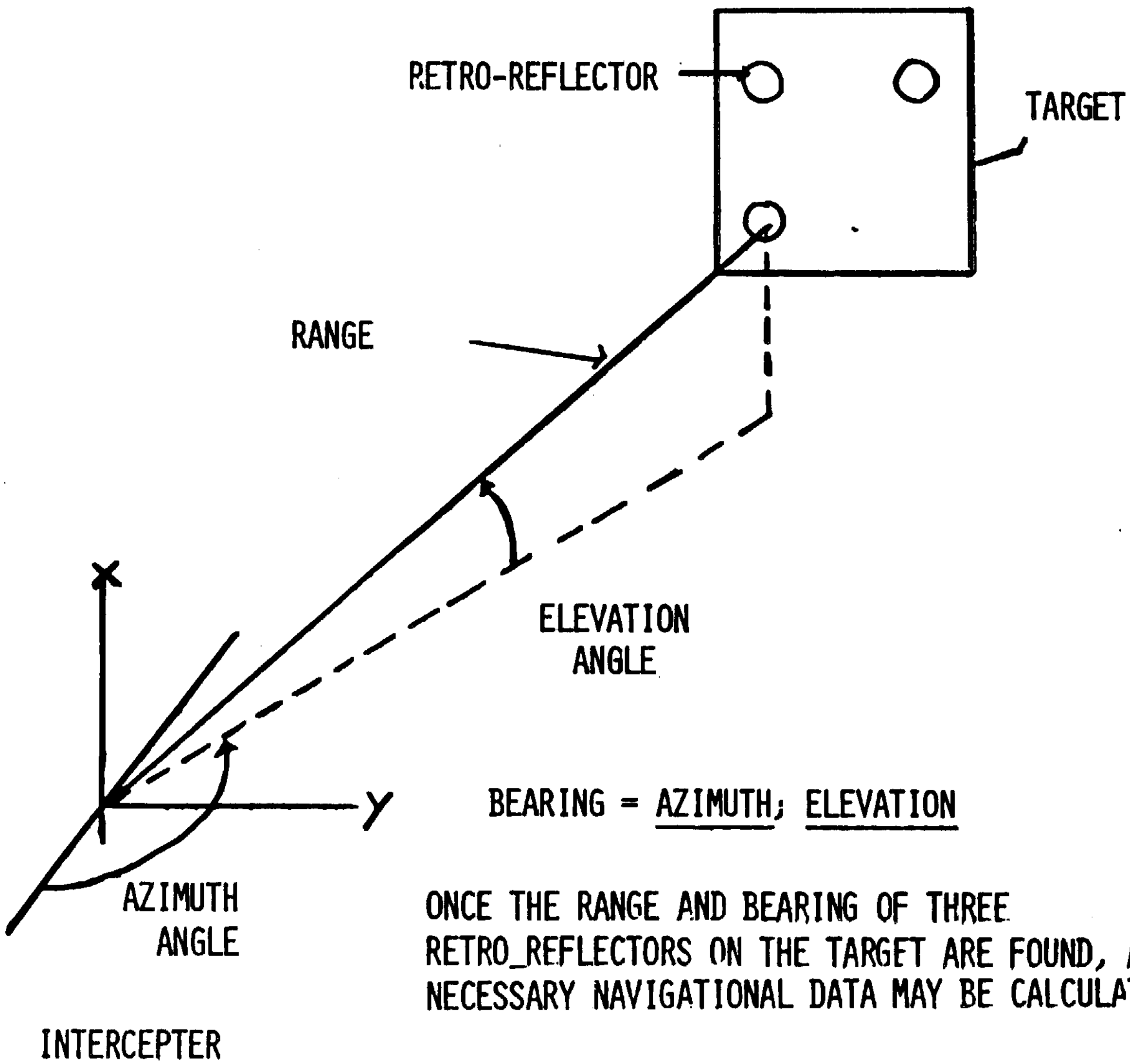
	Presenter: ERWIN/KRISHEN	Date: FEB. 1985

NAVIGATION GEOMETRY



BEARING = AZIMUTH; ELEVATION

ONCE THE RANGE AND BEARING OF THREE RETRO_REFLECTORS ON THE TARGET ARE FOUND, ALL NECESSARY NAVIGATIONAL DATA MAY BE CALCULATED.

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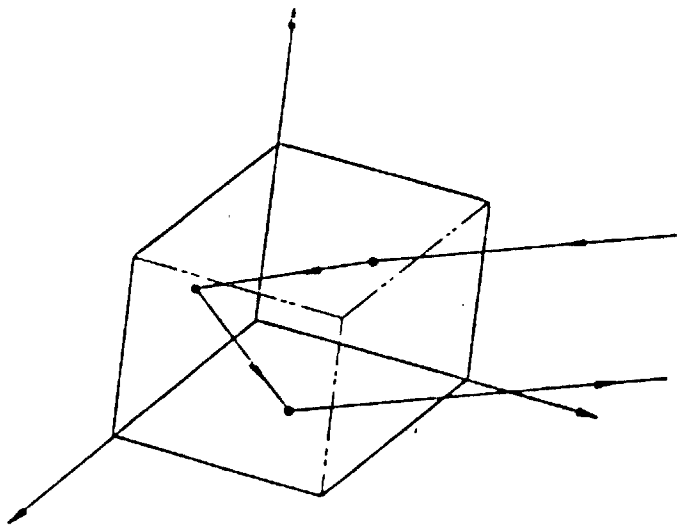
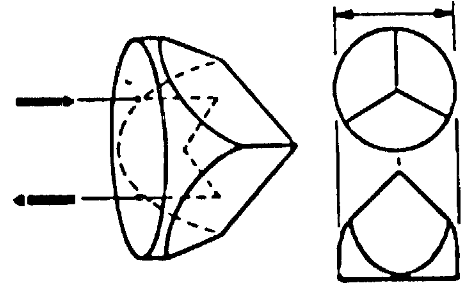
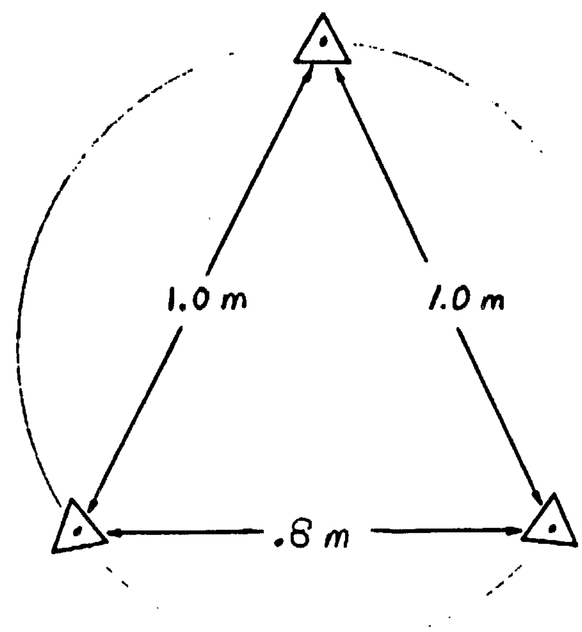
	Presenter: ERWIN/KRISHEN	Date: FEB. 1985

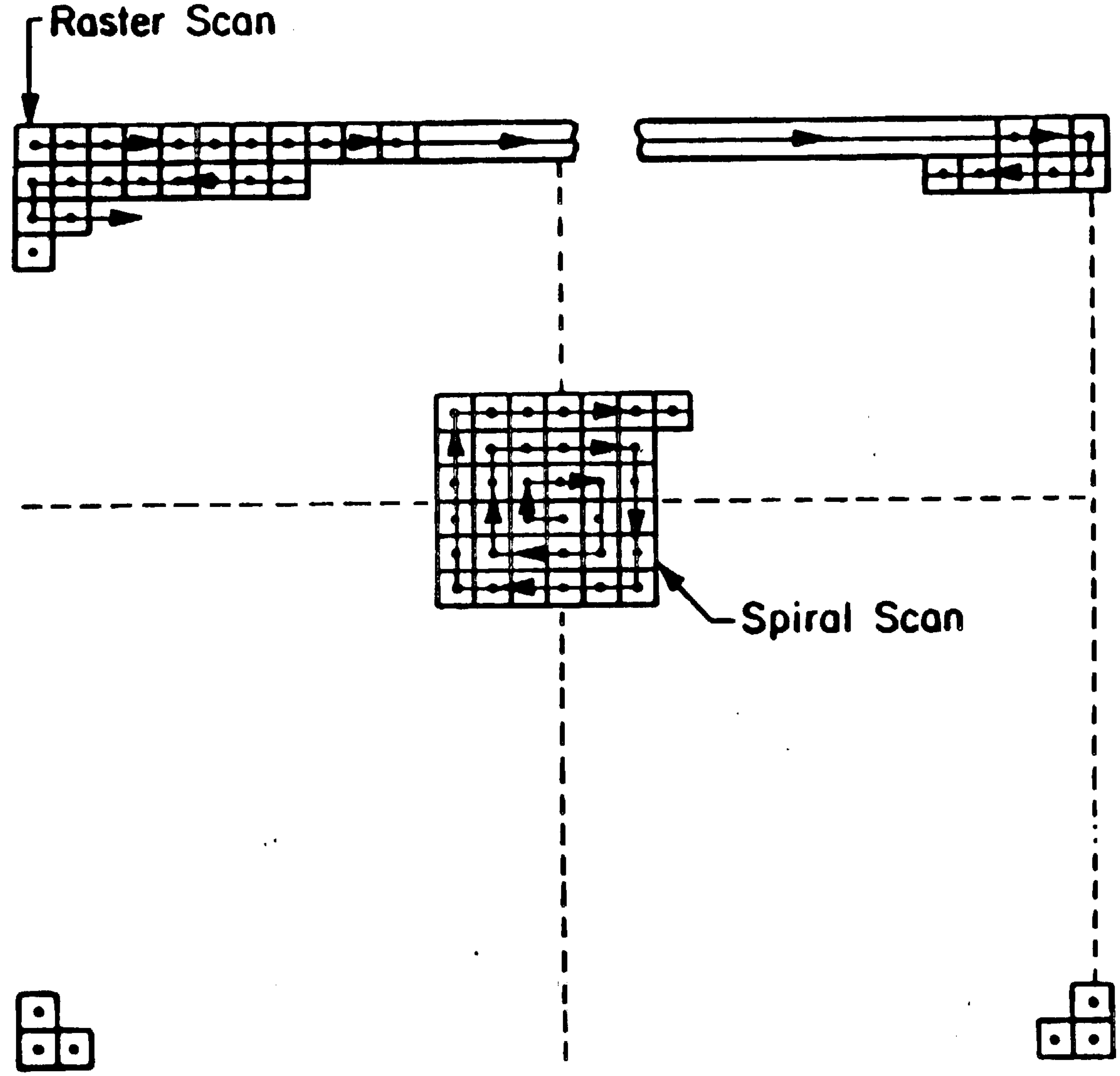




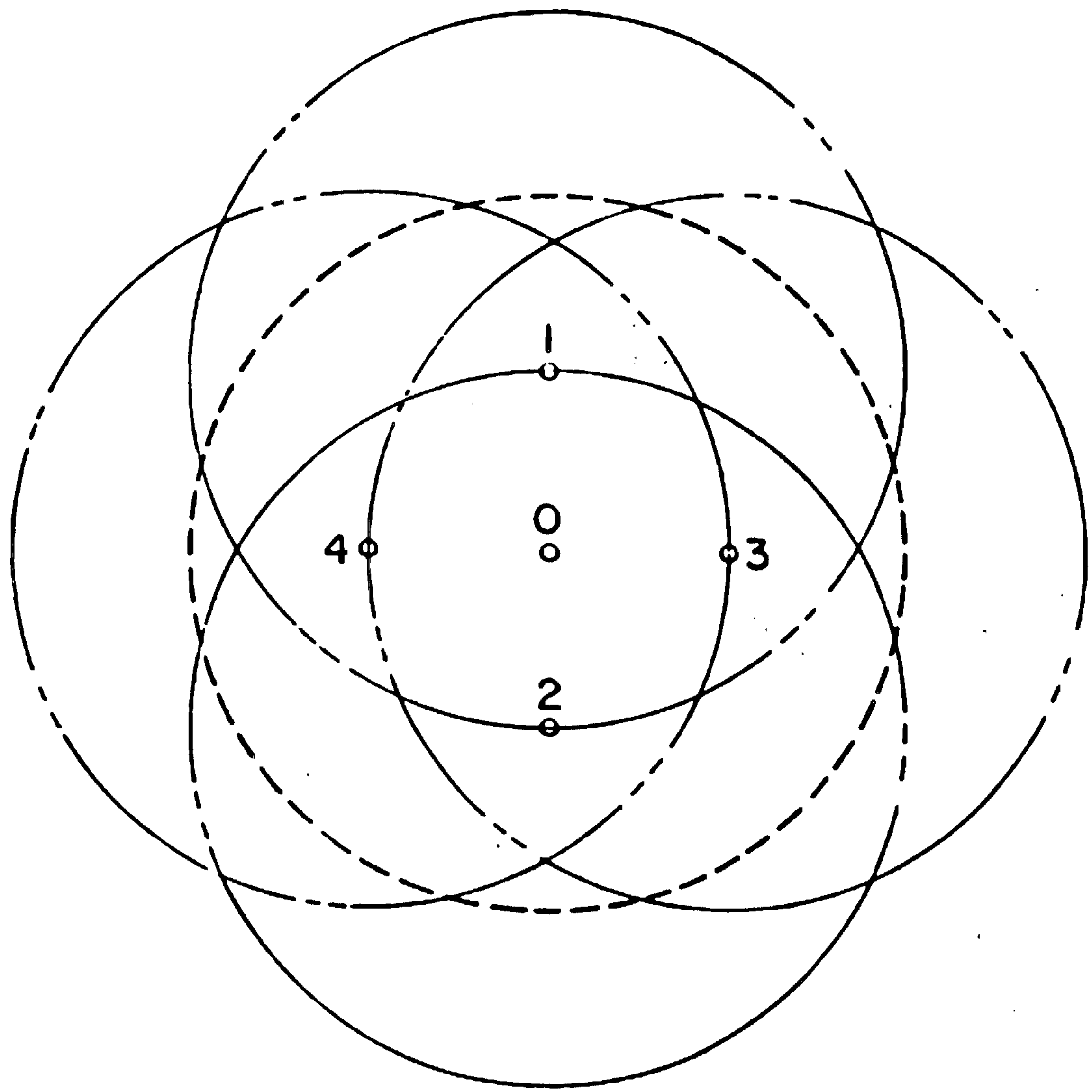


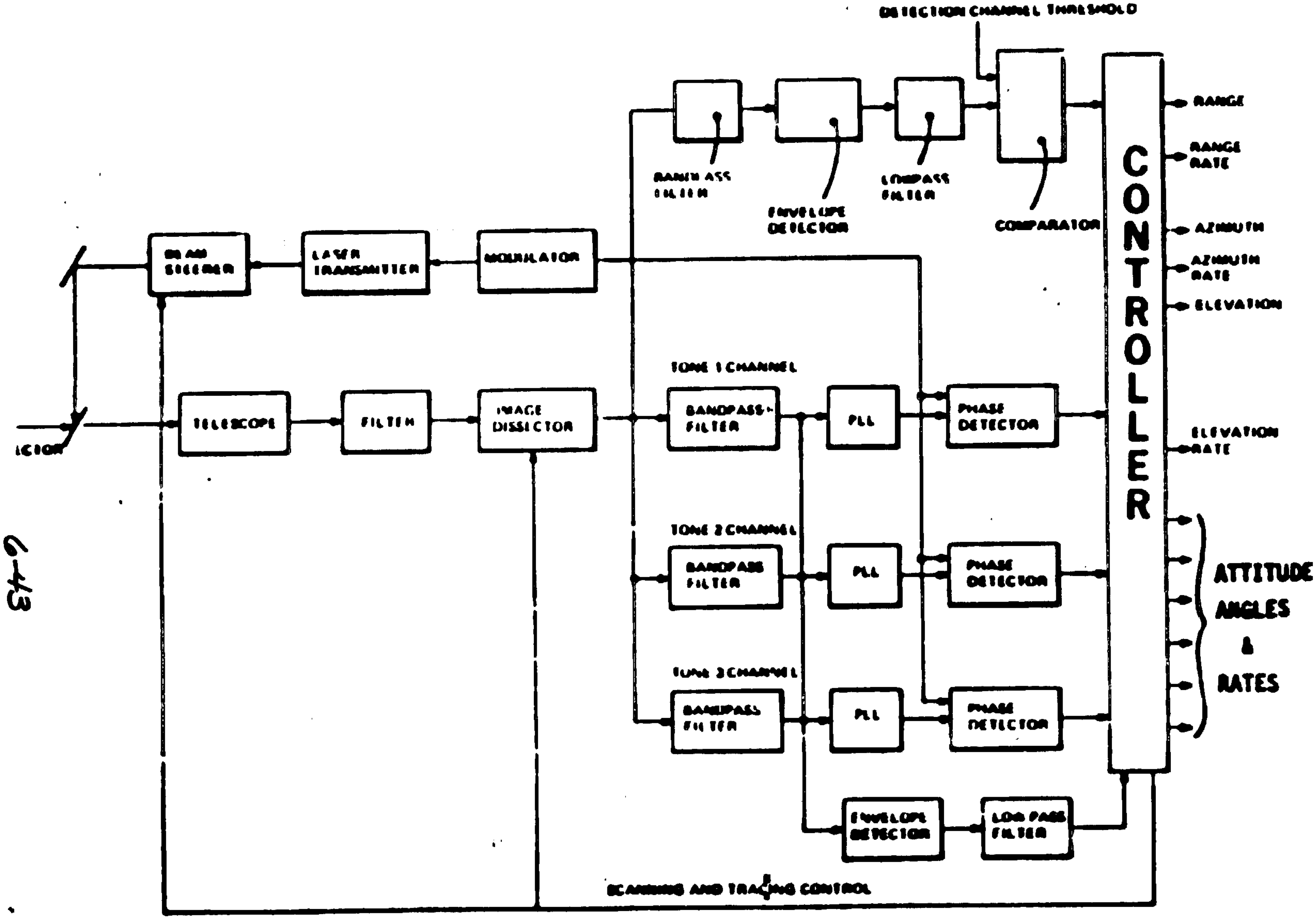
DOCKING AIDS (REFLECTORS)

SEARCH PATTERNS



TRACKING PATTERN
(Sequential Lobing)



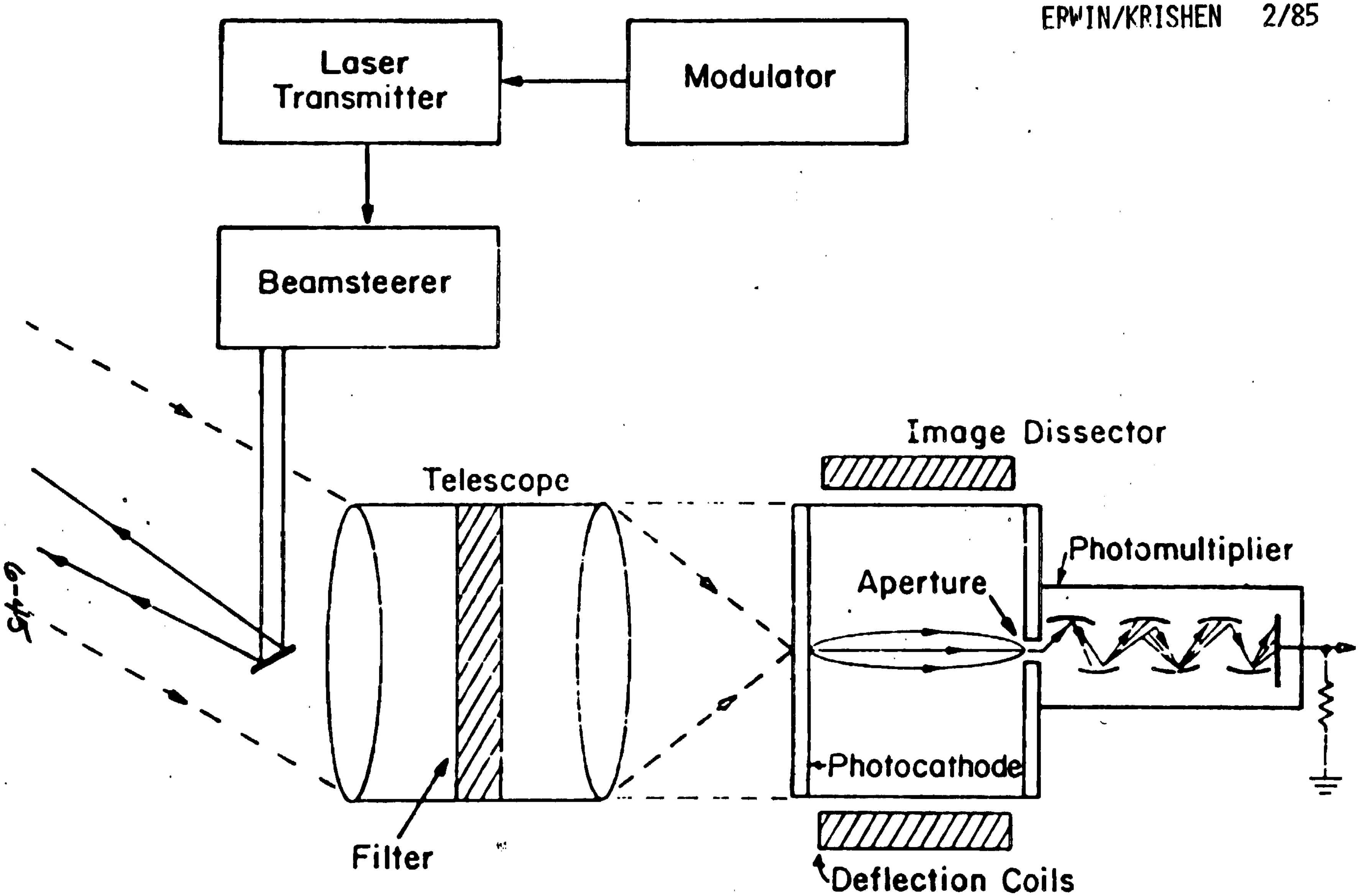


RENDEZVOUS AND DOCKING SENSOR

KEY COMPONENTS

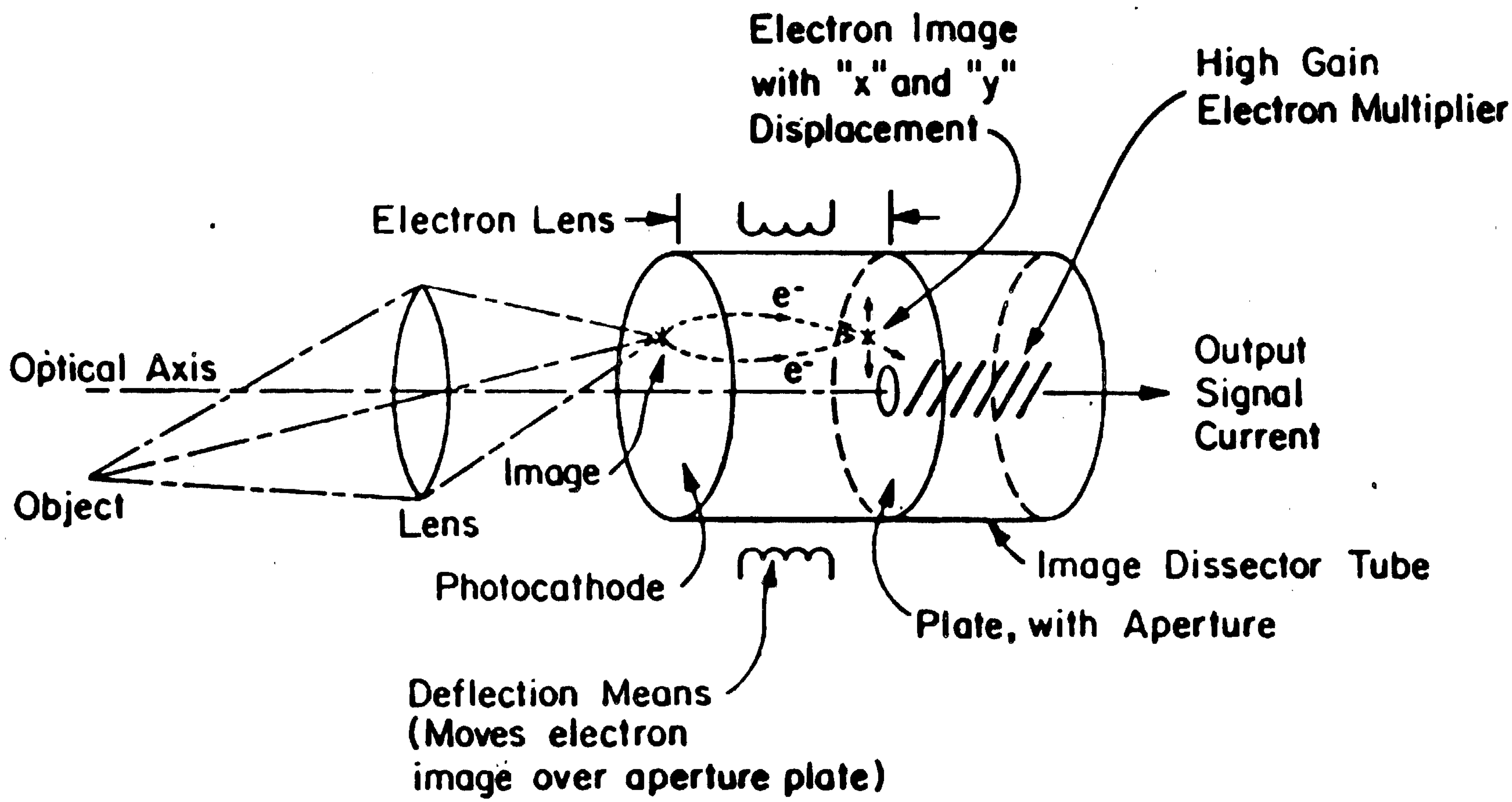
- SEMICONDUCTOR LASERS
- BEAMSTEERERS
- REFLECTORS
- TELESCOPES
- OPTICAL FILTERS
- IMAGE DISSECTORS
- PHASE LOCK LOOPS
- CONTROLLERS

EPWIN/KRISHEN 2/85

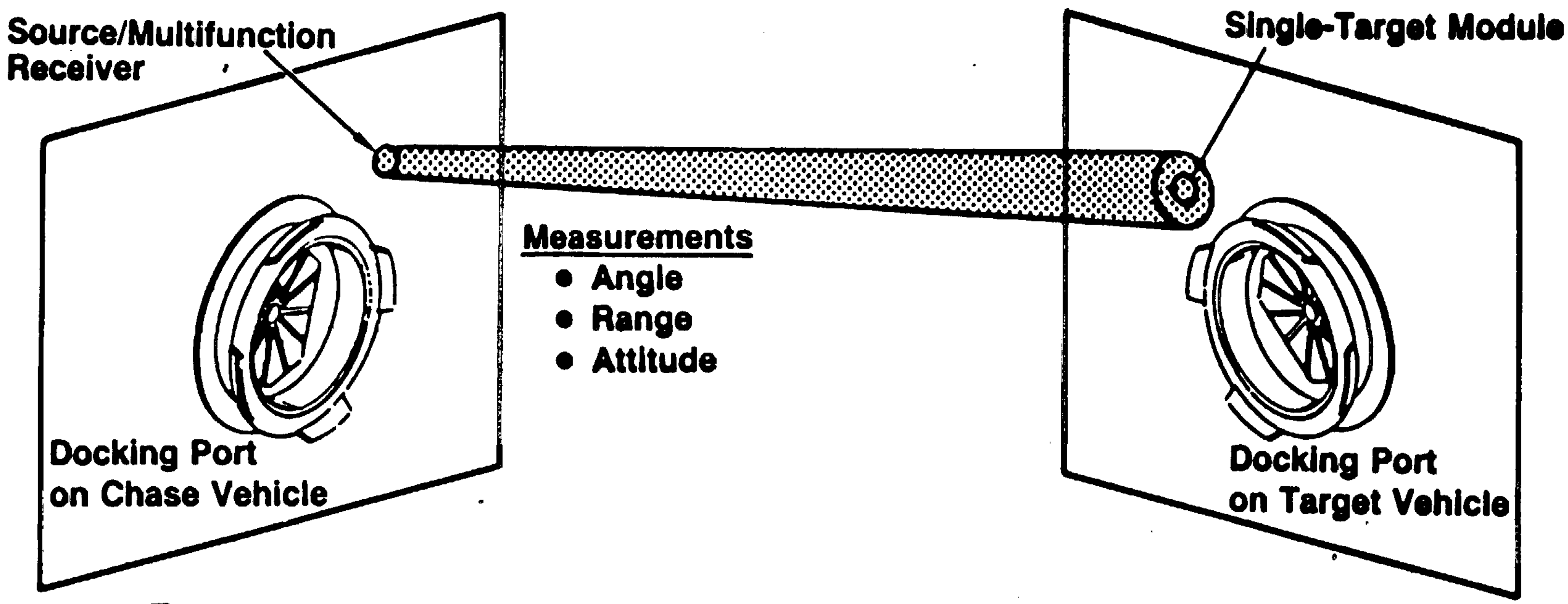


OPTICAL – RADAR FRONT END

IMAGE DISSECTOR



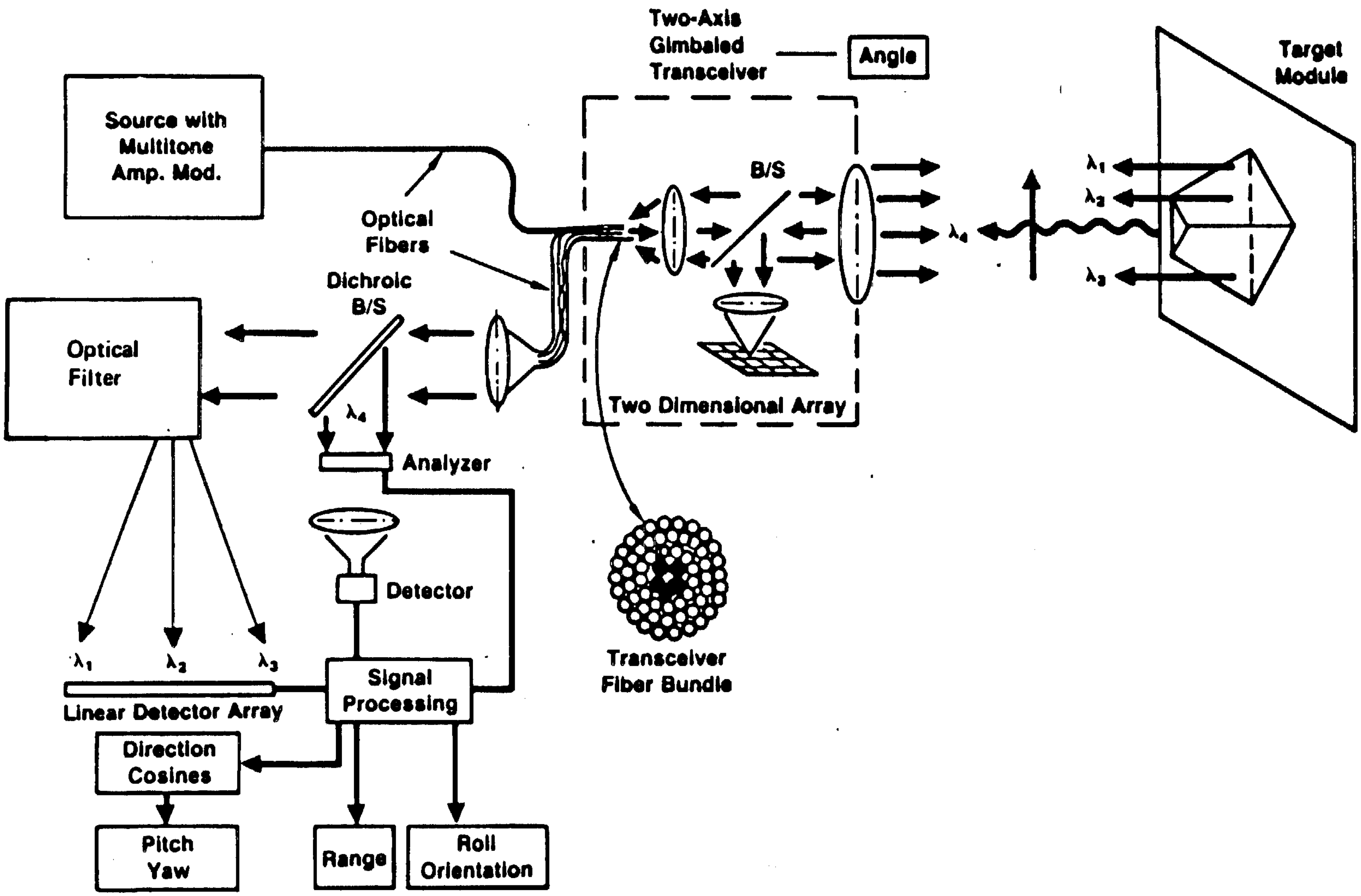
SINGLE TARGET MODULE CONCEPT



Features:

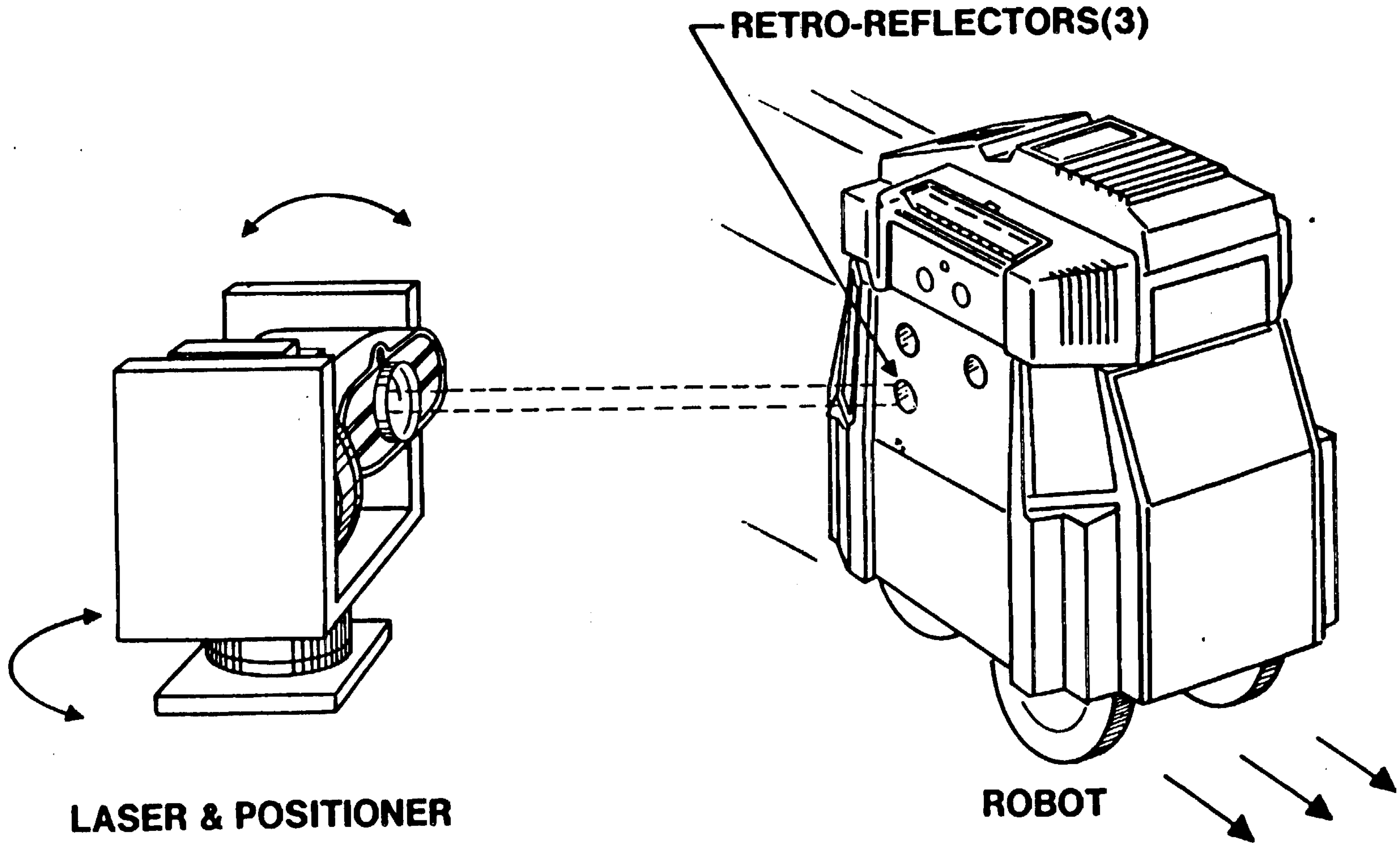
- High Speed, High Accuracy Beam Steerers Not Required
- Two Dimensional Arrays Not Required
- Potential Hardware Simplicity

SINGLE-TARGET MODULE ORIENTATION SENSOR



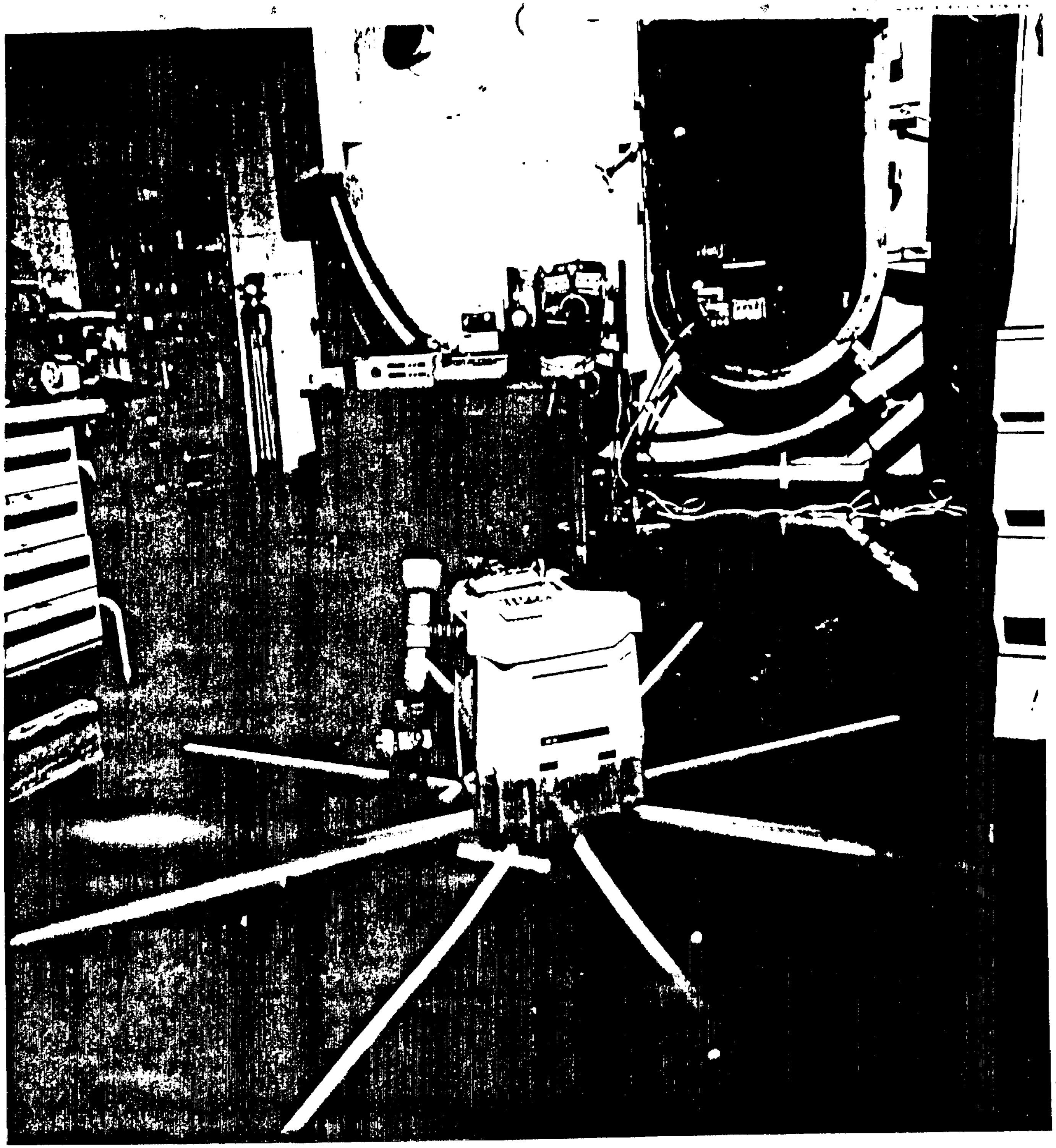
LASER DOCKING SYSTEM

Laboratory Docking Simulation Using Microprocessor Controlled Robot



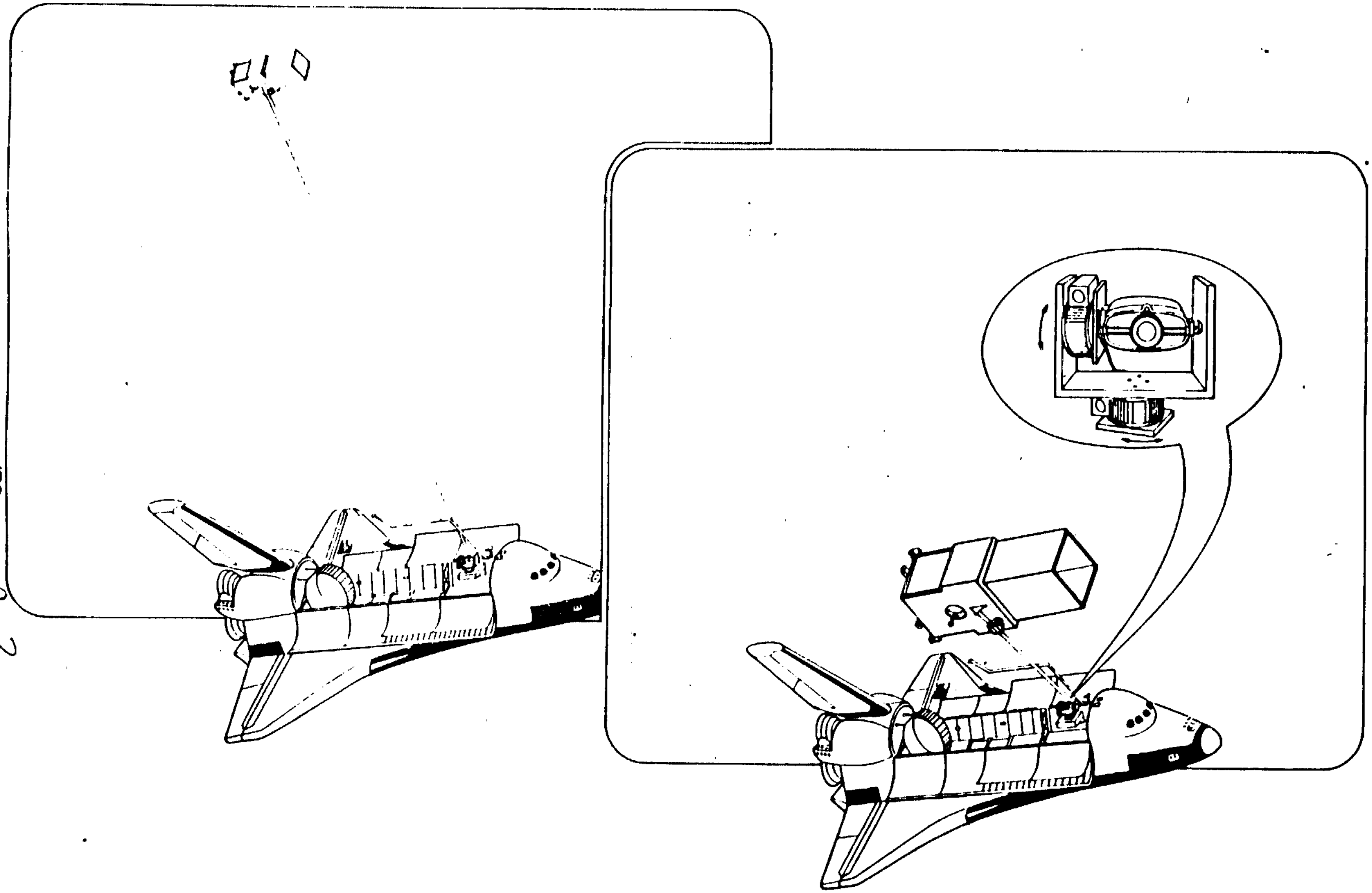
TYPICAL LASER DOCKING SYSTEM LABORATORY TEST RESULTS

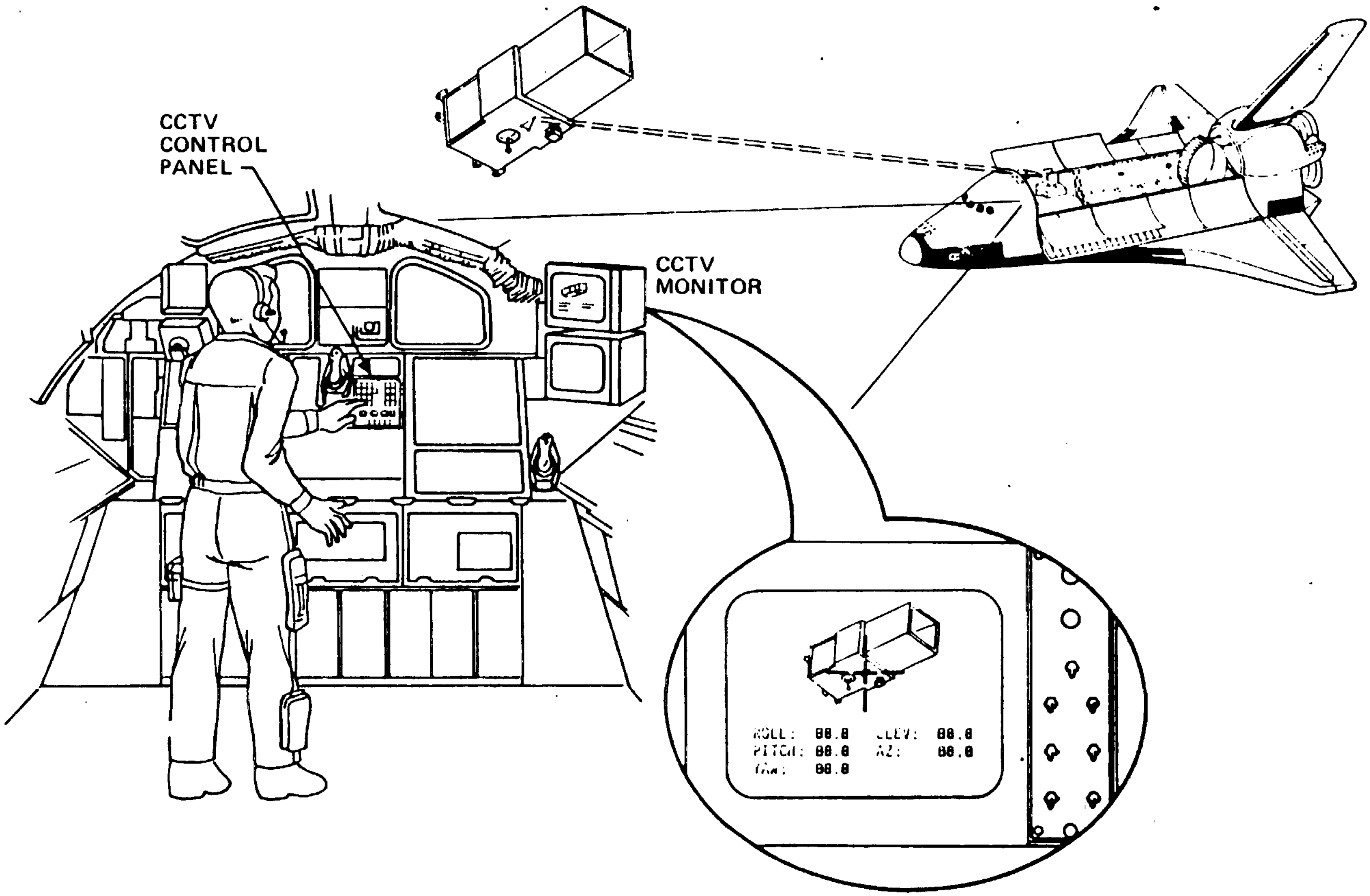
Coverage		
Maximum range	300	meters
Cone angle, radius	20	degrees
Accuracy		
Angle	0.1	degree
Range	0.5	centimeter
Velocity	1.0	cm/sec
Attitude	0.5	degree
Angle output data		
Maximum	20.0	degrees
Resolution	0.01	degree
Word size	12	bits
Rate, maximum	5.0	degrees/sec
Rate, minimum	0.05	degrees/sec
Rate, word size	8	bits
Range output data		
Maximum	300	meters
Minimum	0.002	meters
Rate, maximum	10	meters/sec
Rate, minimum	0.001	meter/sec
Rate, word size	15	bits
Scan		
Horizontal	500	elements
Vertical	500	lines
Receiver		
Lens diameter	0.07	meter
Minimum signal	5.0	nanowatts



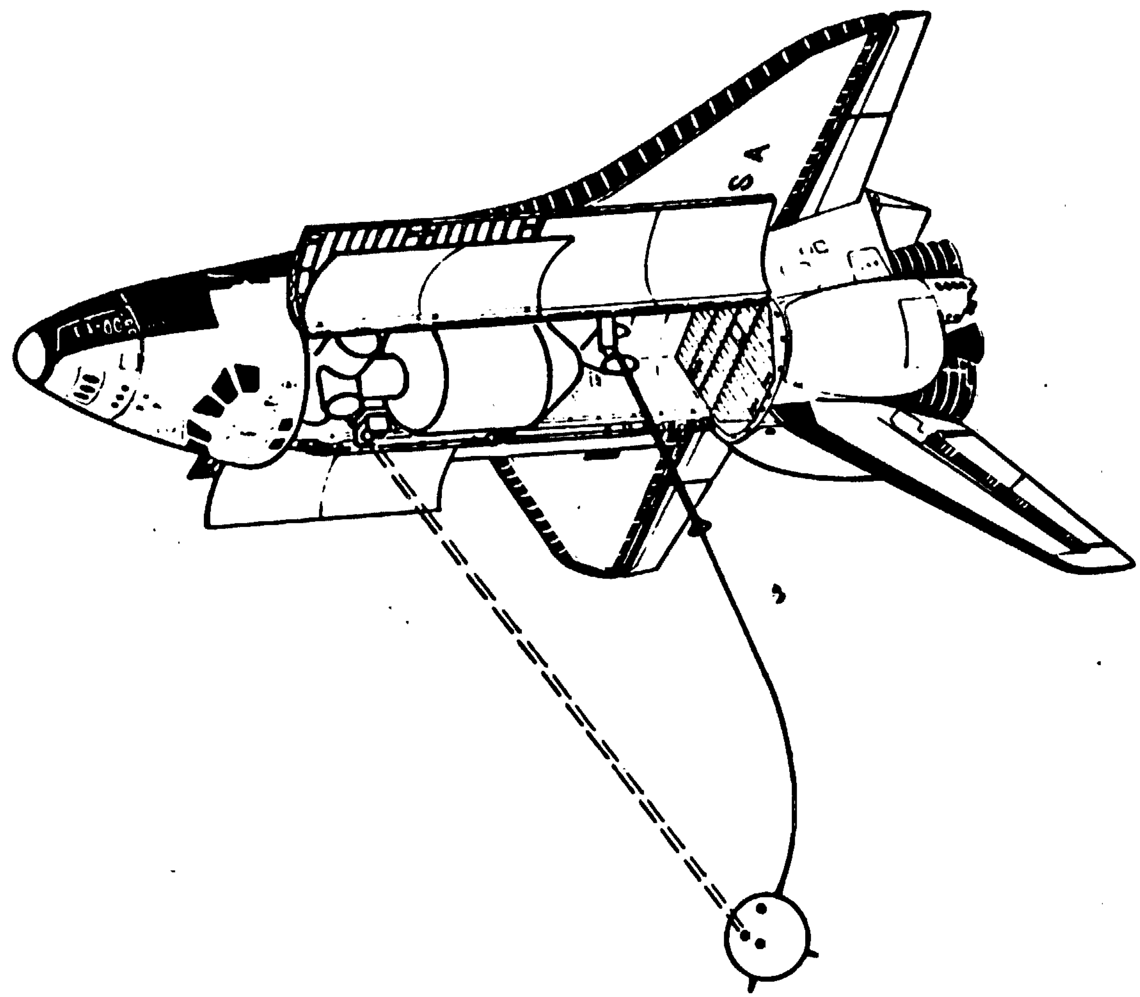
LASER DOCKING SYSTEM LABORATORY SETUP

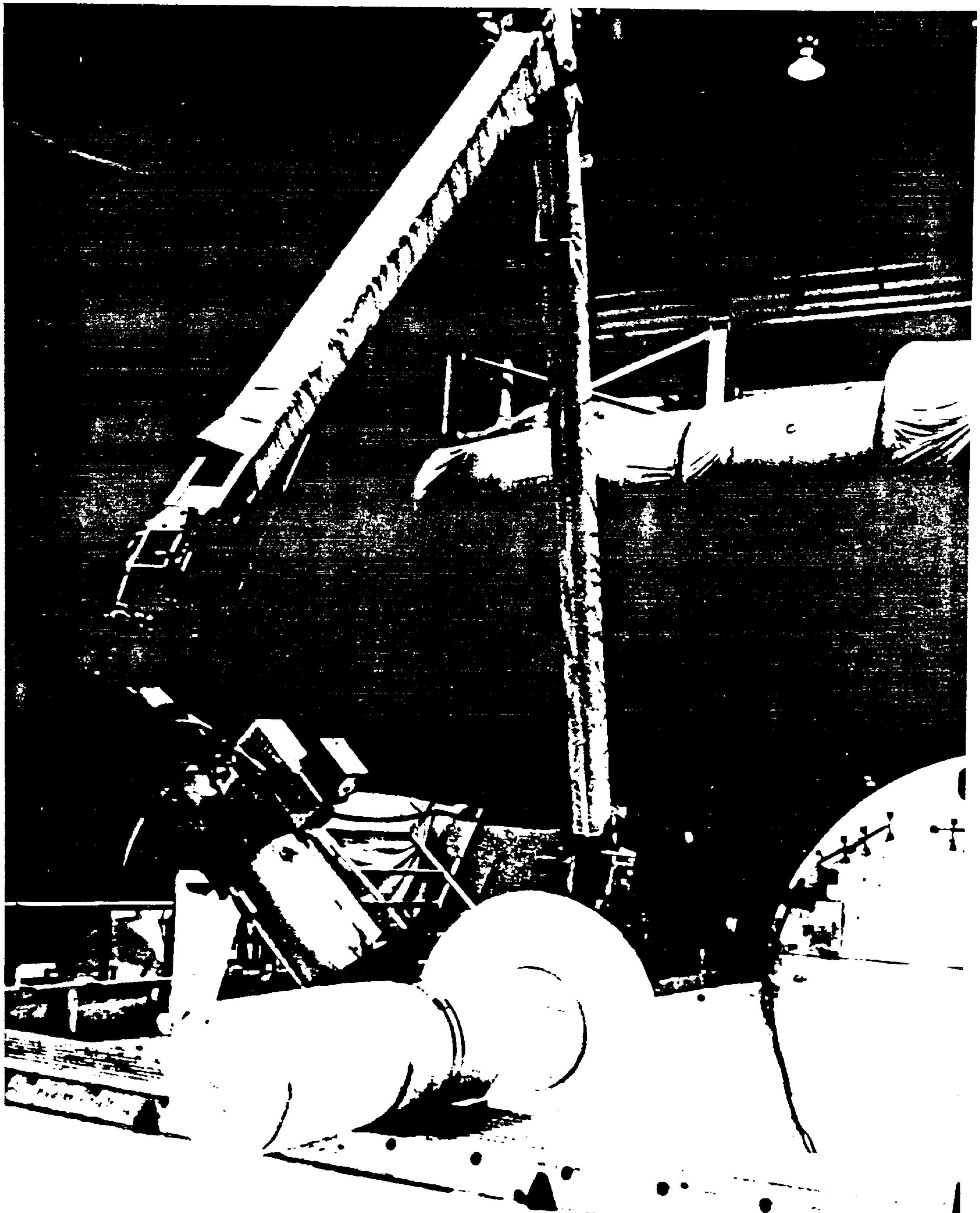
LASER STATIONKEEPING AND DOCKING SENSOR



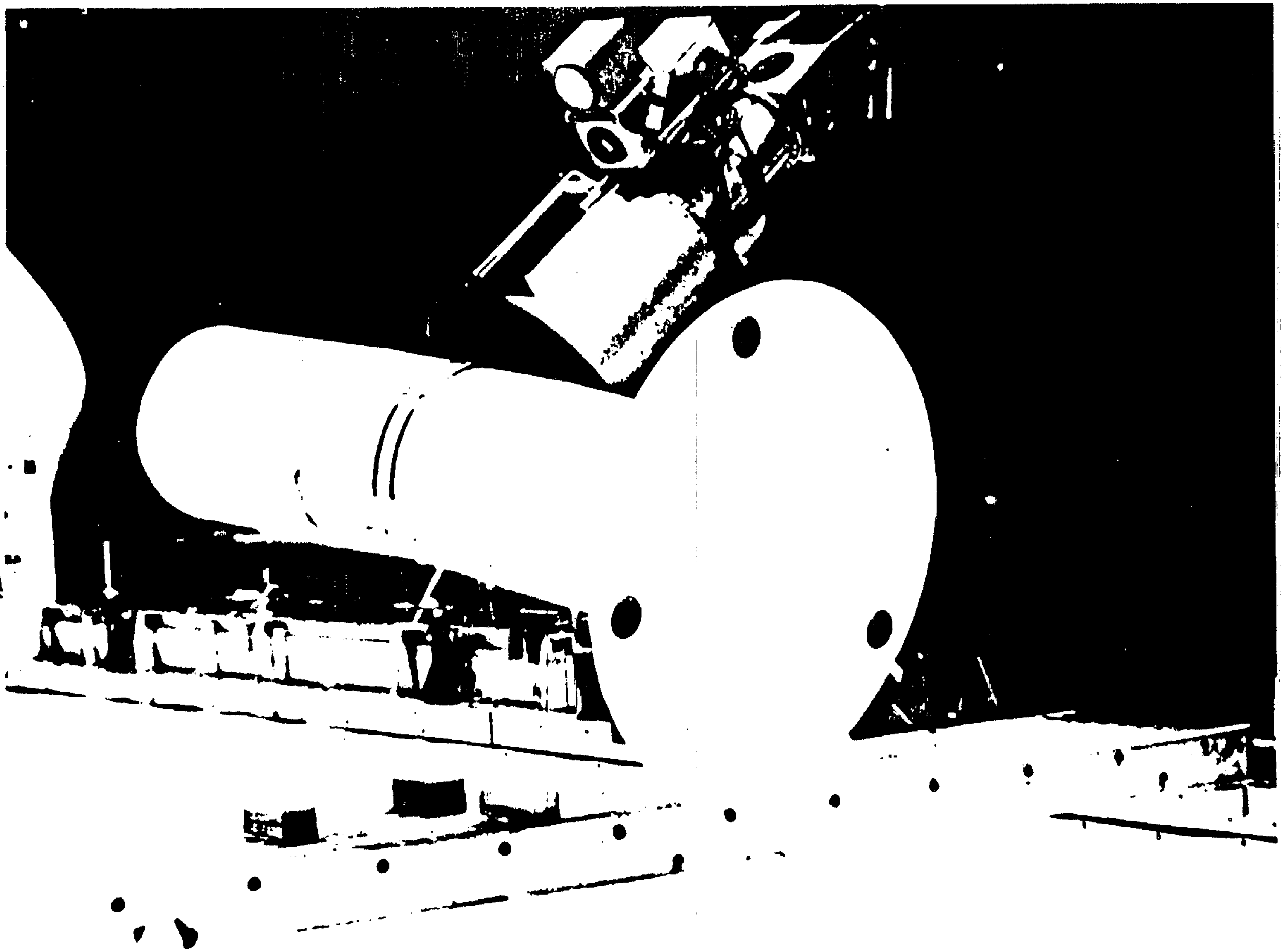

CCTV
CONTROL
PANEL
CCTV
MONITOR

LASER STATION-KEEPING FOR SHUTTLE/TETHERED SATELLITE SYSTEM

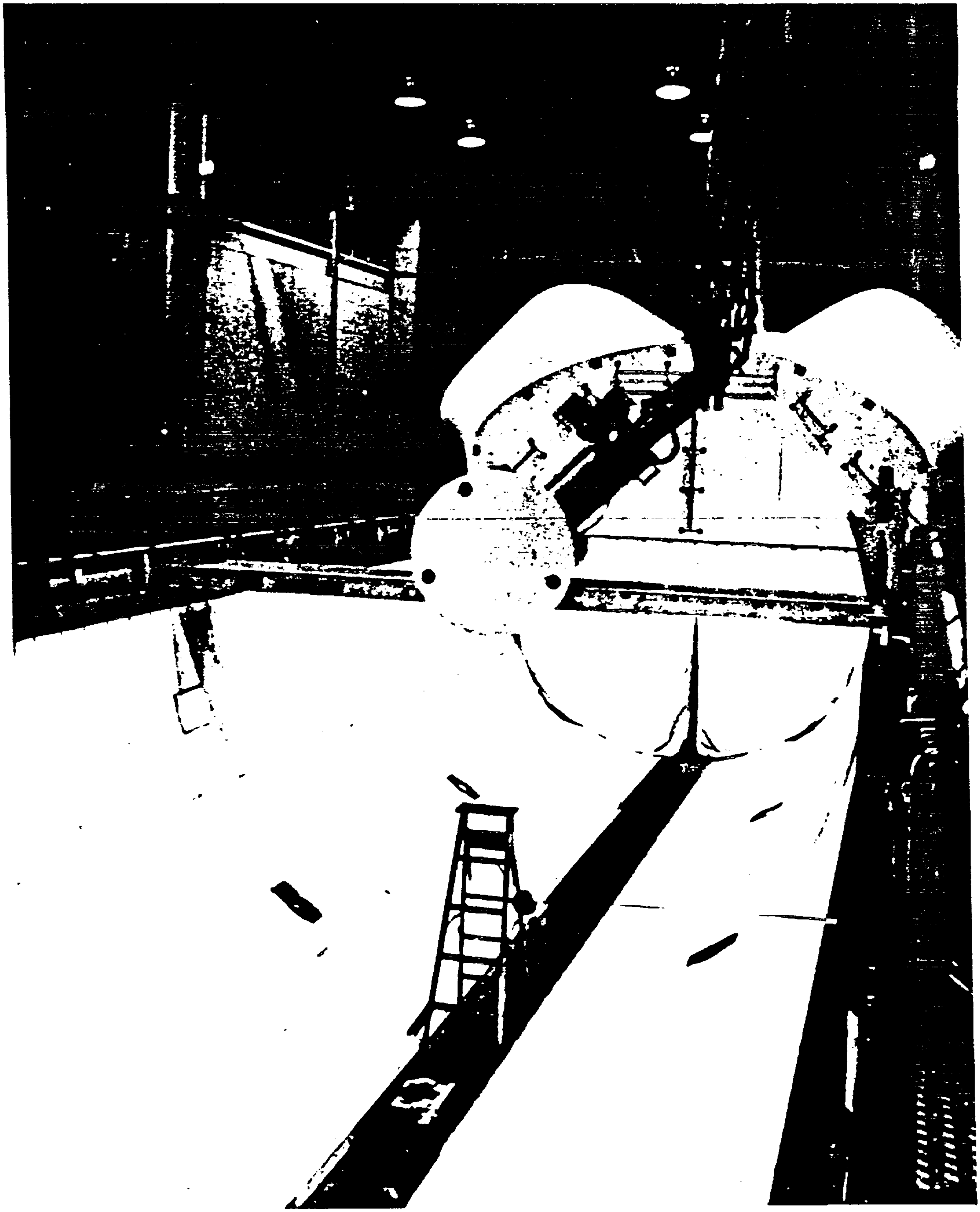




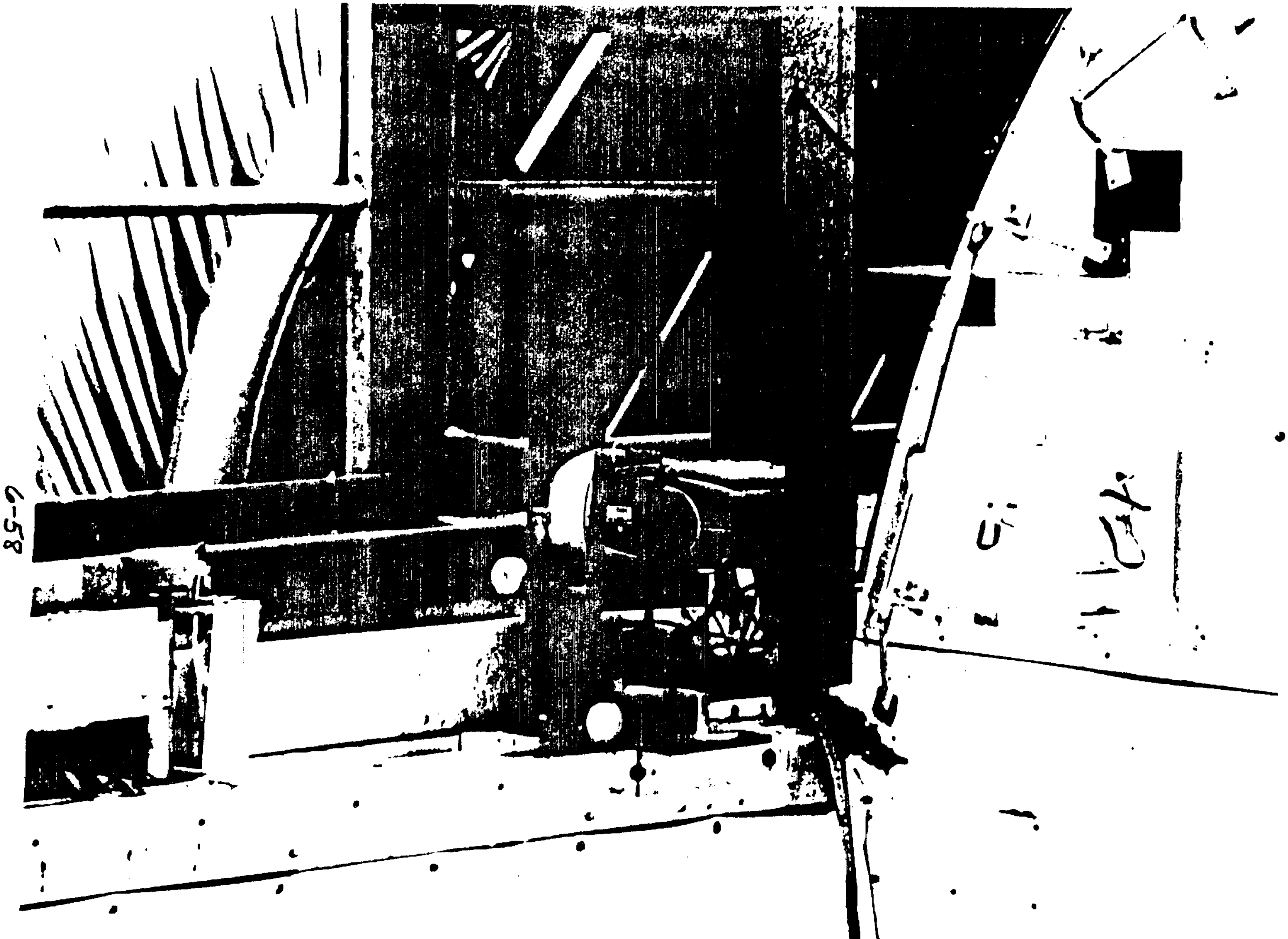
LASER DOCKING DEMONSTRATION IS SET UP IN THE MDF.



TARGET OUTFITTED WITH RETROREFLECTORS IS POSITIONED USING THE MANIPULATOR ARM IN THE MDF (MANIPULATOR DEVELOPMENT FACILITY).

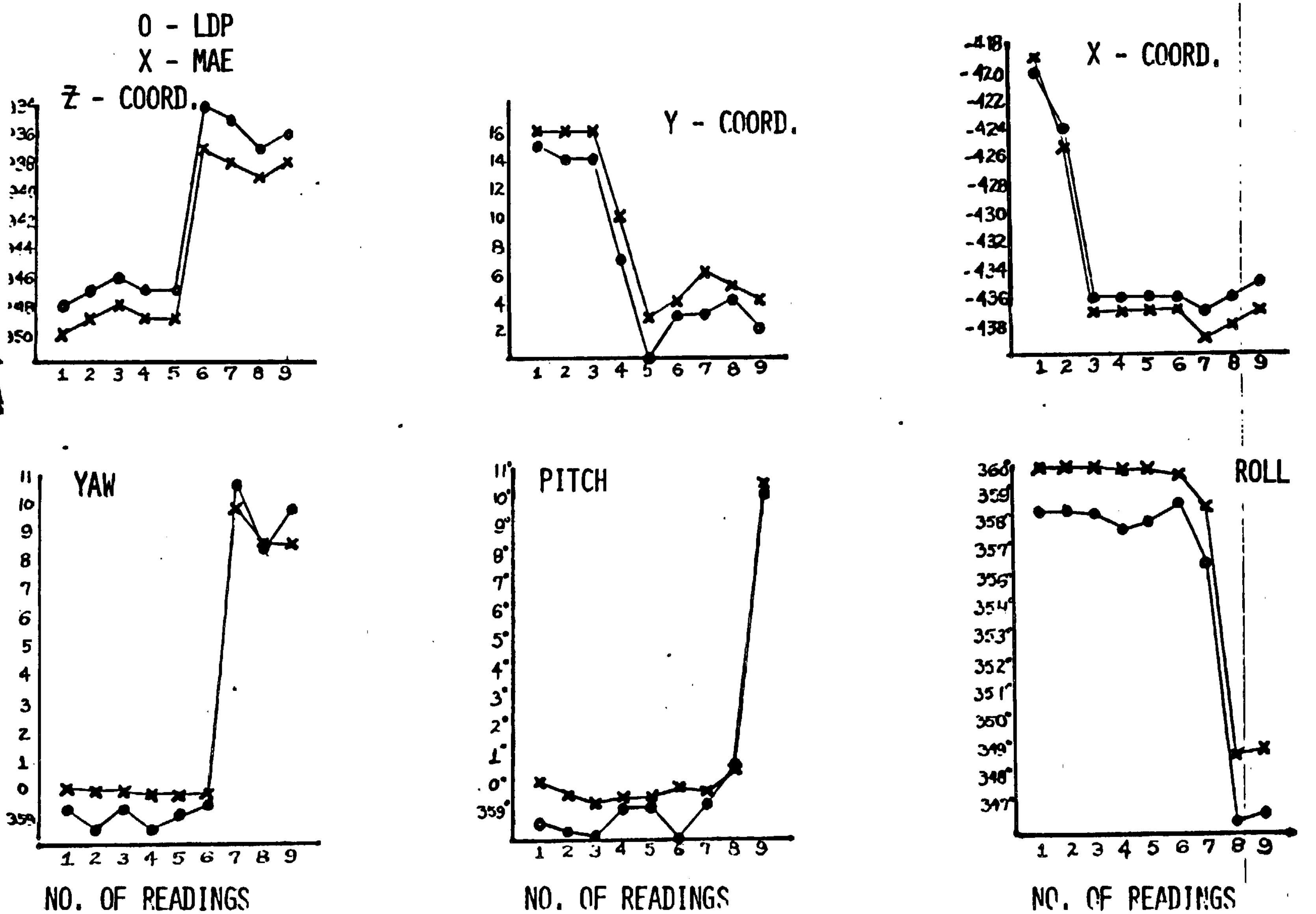


VIEW AS SEEN BY THE LASER SENSOR OF THE TARGET AND PAYLOAD BAY. REFERENCE REFLECTORS CAN BE SEEN ON EITHER SIDE OF THE PAYLOAD BAY.

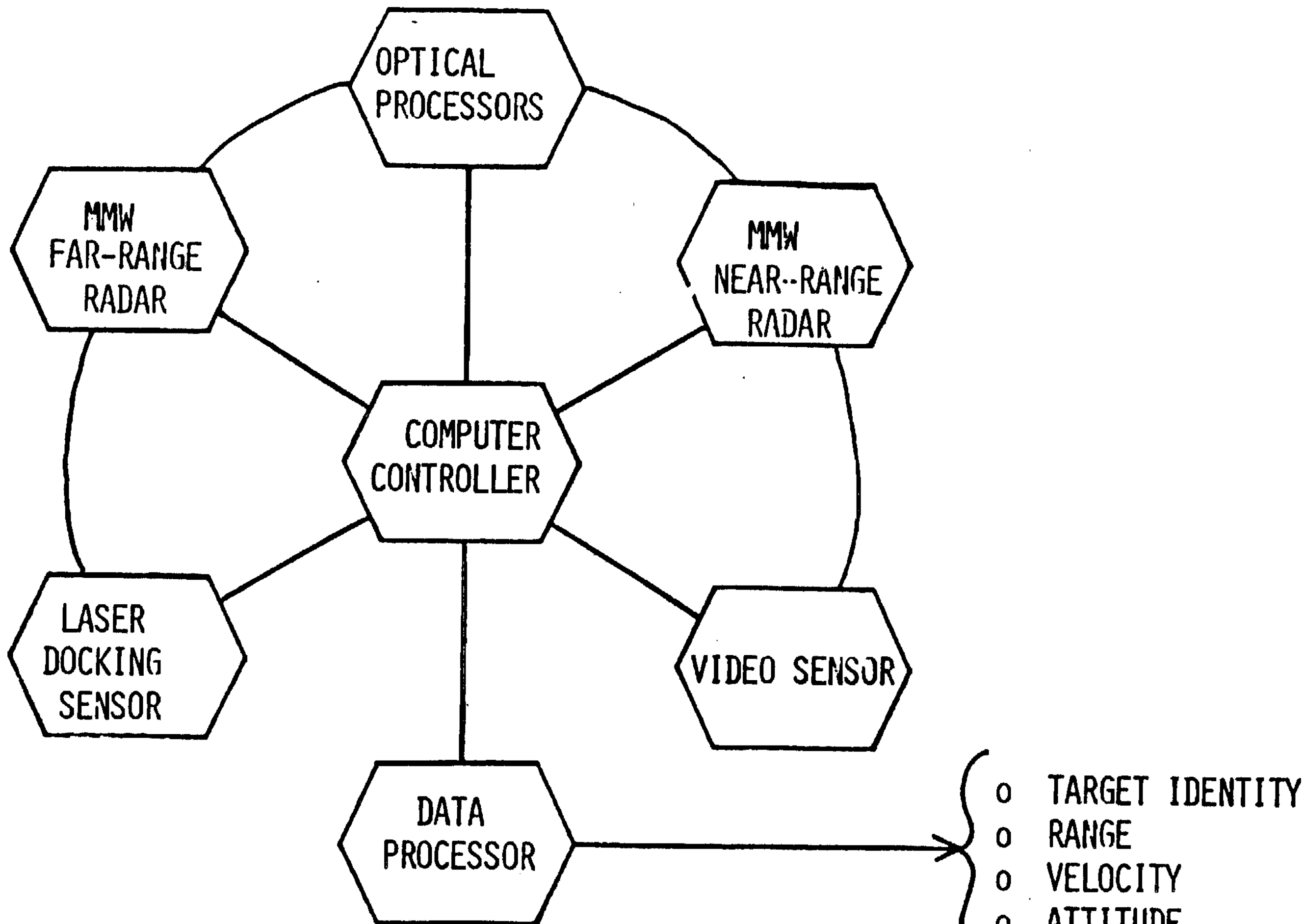


LASER TONE RANGING SENSOR WITH POSITIONER MOTORS IS MOUNTED ON PORT LONGERON NEAR CABIN BULKHEAD IN THE PLDG. 9A MOCK-UP.

COMPARISON OF LASER DERIVED PARAMETER (LDP) WITH THOSE DERIVED FROM MANIPULATOR ARM ENCODERS (MAE)



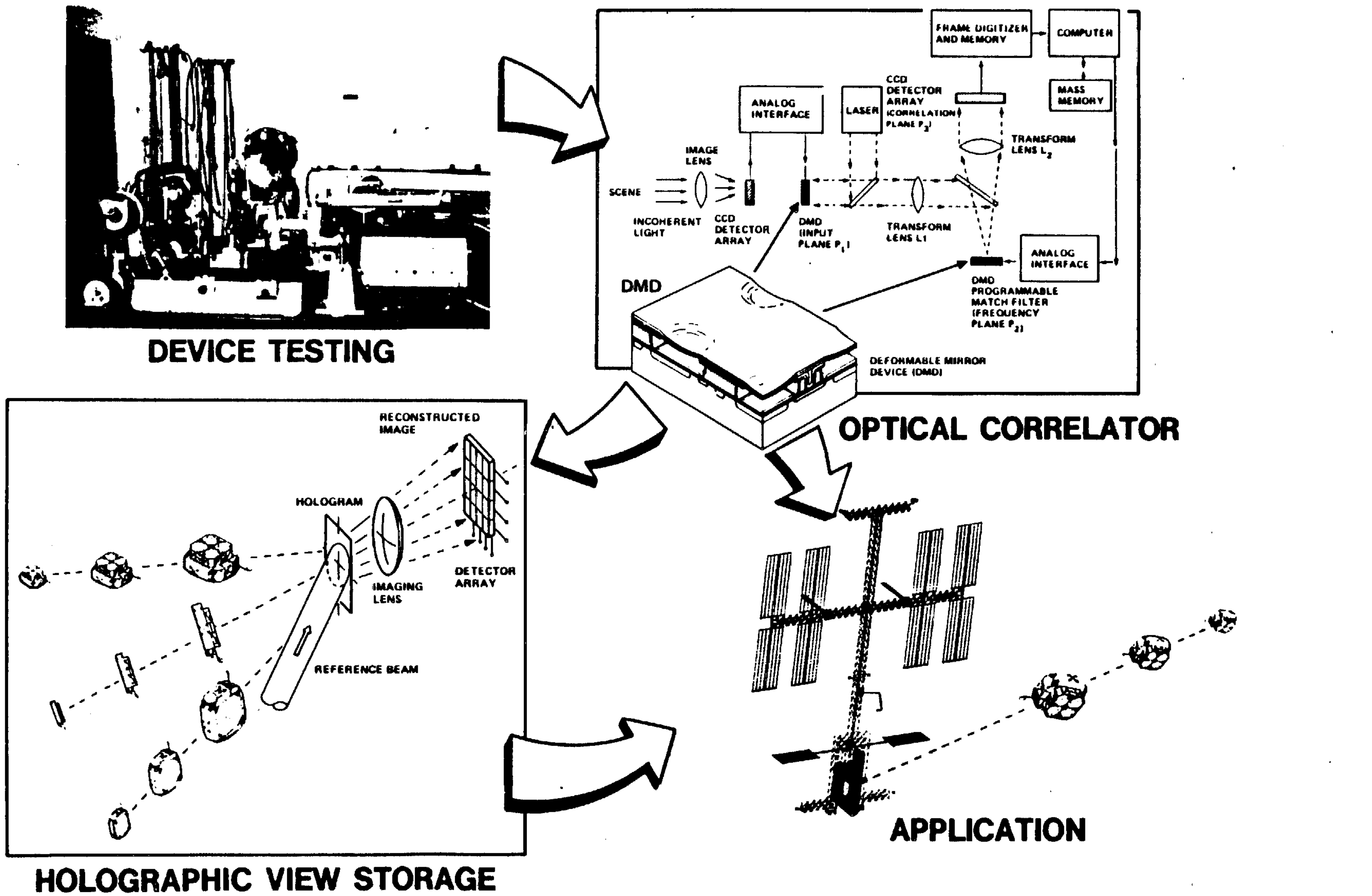
ADVANCED MULTIMODE RAD SYSTEM CONCEPT

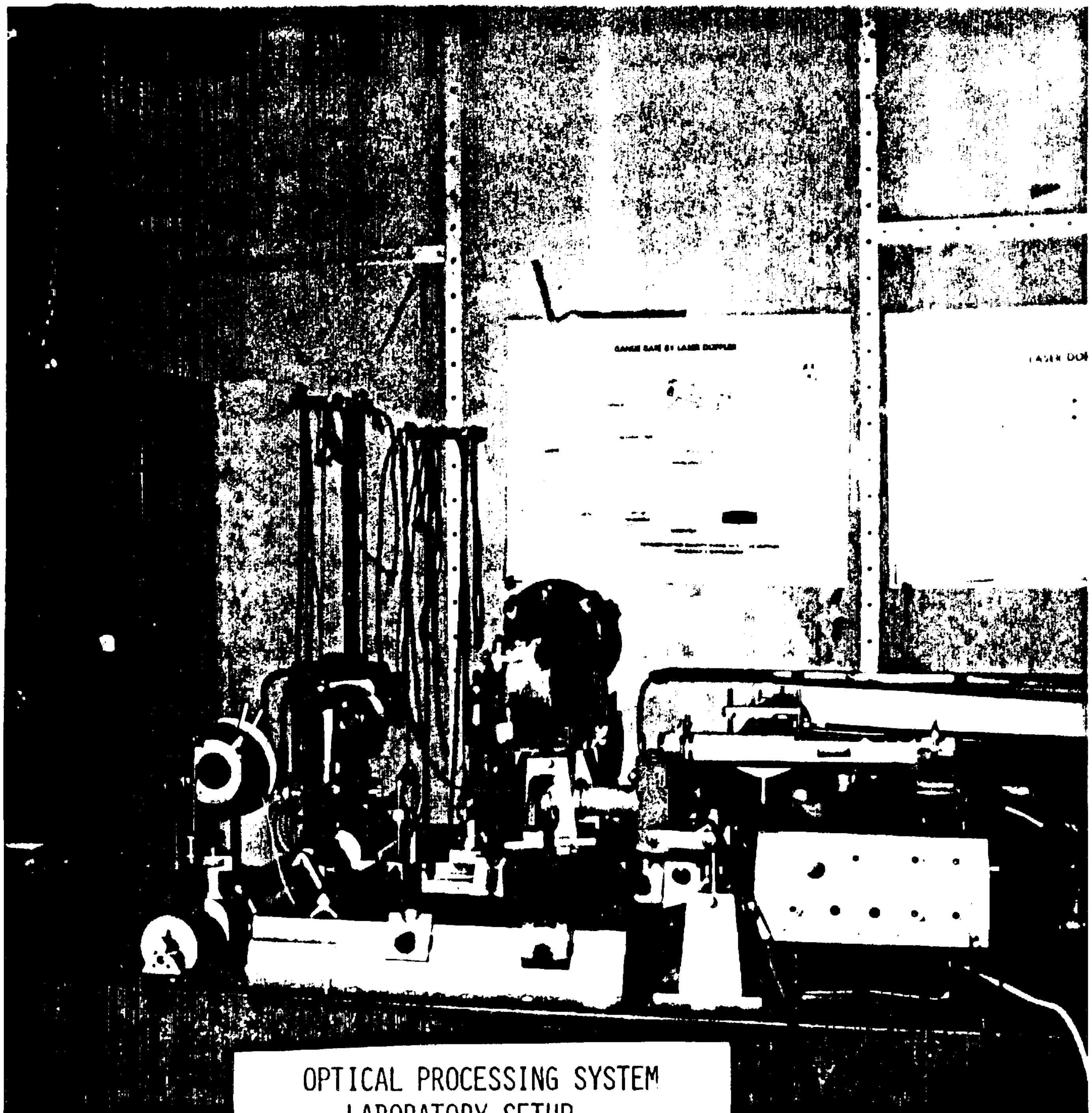


Title	Division/Office	
TECHNOLOGY DEVELOPMENT AREAS	Presenter: ERWIN/KRISHEN	Date: FEB. 1985

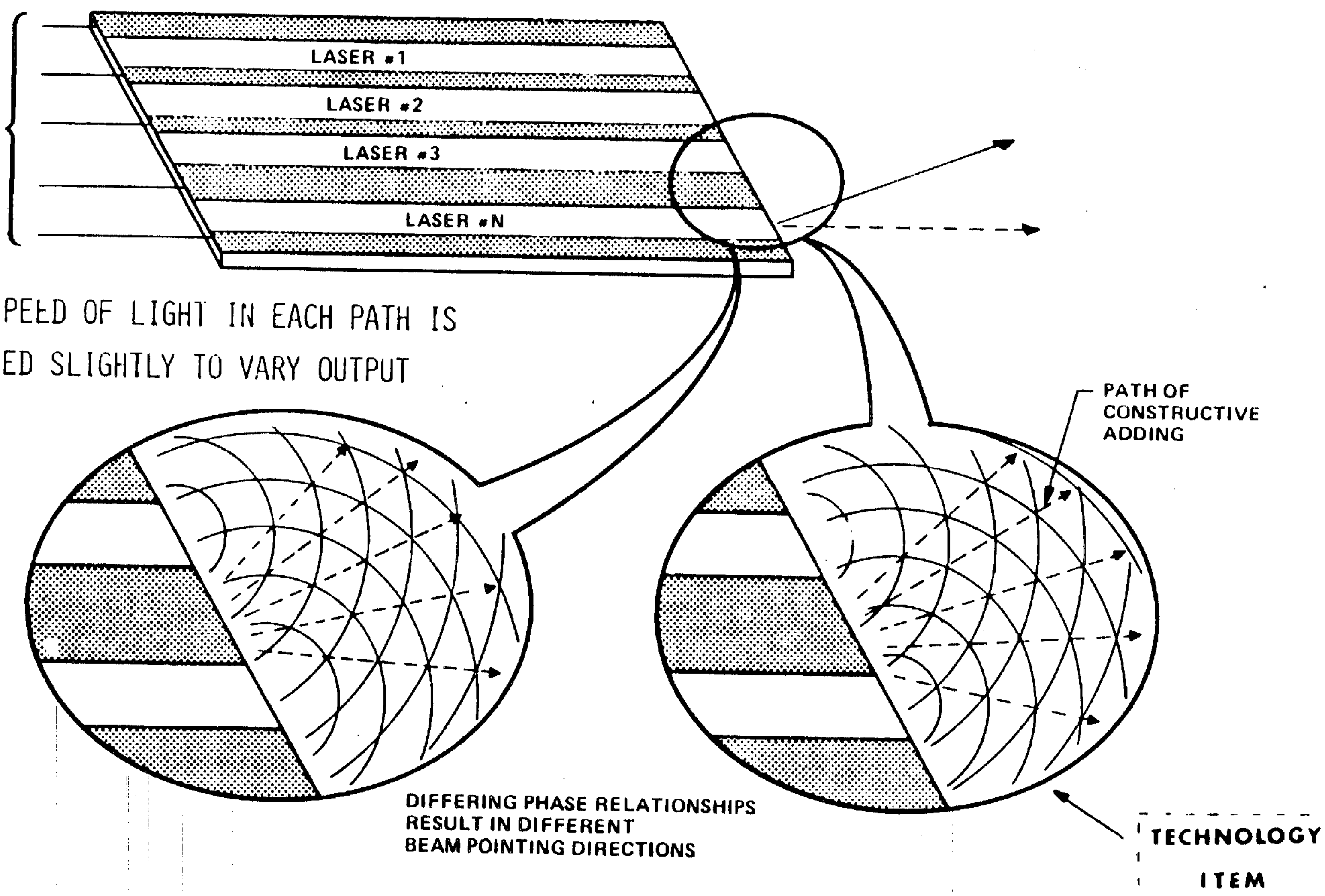
- 0 LASER SYSTEM TECHNOLOGY
  - 0 BEAM STEERING
    - 0 TORQUE MOTORS
    - 0 LASER DIODE PHASED ARRAYS
  - 0 CID DETECTOR ARRAYS
    - 0 256 X 256 ARRAY
    - 0 SOFTWARE TECHNIQUES TO GIVE RESOLUTION OF 1 PART IN 25,000
  - 0 APD DETECTOR WITH ROTATING SLIT
    - 0 TONE RANGING THROUGH APD DETECTOR
    - 0 MEASURE RADIAL DISTANCE AND ANGULAR POSITION OF SPOT THROUGH SLIT
  - 0 IMAGE DISSECTOR RECEIVER
    - 0 ENHANCED RESPONSE OF IMAGE DISSECTOR TUBE IN THE INFRARED SPECTRUM TO ALLOW USE OF SEMICONDUCTOR LASER SOURCE.
  - 0 MM WAVE COMPONENTS/SUBSYSTEMS
    - 0 40 GHZ TO 200 GHZ
    - 0 DISTRIBUTED PHASED ARRAY ANTENNAS
    - 0 SOLID STATE SOURCES AND OTHER DEVICES
    - 0 MMIC MIXERS
    - 0 LOW NOISE RECEIVERS/AMPLIFIERS

PROGRAMMABLE MASK TECHNOLOGY




OPTICAL PROCESSING SYSTEM
LABORATORY SETUP

OPTICAL PHASED ARRAYS



- THE SPEED OF LIGHT IN EACH PATH IS CHANGED SLIGHTLY TO VARY OUTPUT PHASE

ACKNOWLEDGEMENT

THE AUTHORS WOULD LIKE TO ACKNOWLEDGE THE CONTRIBUTIONS OF RAY NUSS, DAN O'CONNELL, AND PENNY SAUNDERS OF NASA/JSC, DR. SAM STEPHAN OF LOCKHEED CORPORATION, AND WILL OTAGURO OF MCDONNELL DOUGLAS TO THIS PAPER.

N.A.S.A.

Rendezuous and Proximity Operations Workshop
February 19 - 22, 1985
Lyndon B. Johnson Space Flight Center
Houston, Texas 77058

Session 6
Laser and R.F. Radar Systems and Technology

Paper 6 - 2
Highly Stable Nd:YAG Laser Technology
For Range and Range Rate Measurements

by

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April, 1985

HIGHLY STABLE Nd:YAG LASER TECHNOLOGY FOR RANGE AND RANGE RATE MEASUREMENTS

Robert L. Byer

INTRODUCTION

In his classic text "Introduction to Radar Systems"[1], M.I. Skolnik describes the subject matter of engineering as components, techniques and systems. Components are the building blocks, that with proper technique, yield a system.

In this paper I describe recent progress toward the development of solid state lasers and components that with the techniques of Radar, so eloquently described by Skolnik, will yield a coherent Radar system at 1 micrometer wavelength or at a frequency of 300 THz or 3×10^{14} Hz.

The work has been motivated by the recognition that coherent Radar at 1 µm may offer system measurement advantages compared to 10 µm based Radar for remote sensing of wind.[2] Further, an all solid state transmitter based on laser diode pumped solid state lasers such as Nd:YAG are potentially long lived, compact, efficient sources of coherent radiation that are also compatible with the system constraints of a free-flying, 800 km altitude, orbiting satellite that is required for global wind field measurements.[3]

SOLID STATE LASER COMPONENTS

The components required for the construction of a coherent Radar system at 1 micrometer wavelength include a frequency stable, narrow bandwidth oscillator, an isolator, a high gain pre-amplifier and power amplifier, beam splitters, optical beam routing methlds and a detector.

At the outset of our program the most critical components that at that time had not been demonstrated, were the narrow bandwidth coherent oscillator and the high gain amplifier. Beam routing technology based on single mode, polarization preserving optical fiber was also under development and not yet available.

Our first goal was to demonstrate that Nd:YAG oscillator technology was adequate for wind measurements. The requirement for wind speed measurements is 1m/sec or 1 MHz Doppler shift at 1 μm wavelength.

We set out to demonstrate a coherent Nd:YAG oscillator with less than 1 MHz linewidth for a 5 msec time interval which is the round trip time of optical propagation from an 800 km orbiting satellite. Our efforts yielded a 100 kHz linewidth, flashlamp pumped, feedback controlled Nd:YAG oscillator system. The results were published in 1982[4] and gave encouragement to the prospect that Nd:YAG technology could form the basis for coherent Radar at 1 μm wavelength.

During the development of the flashlamp pumped, water cooled, Nd:YAG oscillator, it occurred to us that the design was overly complex for a low power local oscillator. Progress

in laser diodes led us to investigate the feasibility of pumping small Nd:YAG oscillators with diode lasers.

Early results showed 1 MHz linewidths for dye laser pumped, 5 mm long monolithic Nd:YAG oscillators. The amplitude fluctuations inherent in the cw dye laser source led to frequency fluctuations in the Nd:YAG oscillator that were well in excess of the predicted Shawlow/Townes linewidth of near 1 Hz per milliwatt of laser output power.

Efforts to laser diode pump the monolithic Nd:YAG oscillator crystal were finally successful in 1984. Initial results led to improvements in the optical system for imaging the laser diode pump radiation into the TEM_{00} mode volume of the Nd:YAG crystal. The use of single mode diode lasers and a self-foc lens for imaging proved to be both simple and effective. Nd:YAG oscillator thresholds as low as 1 mW were achieved. Slope efficiencies of 25% were demonstrated.

The key improvement was a reduction in laser linewidth to less than 10 kHz.[5] The linewidth reduction was a result of the power stability of the laser diode pump source and the monolithic design of the Nd:YAG oscillator.

In these early experiments the laser linewidth was measured by mixing the output in a photodector of two independent Nd:YAG oscillators set on a table in an open laboratory environment. In the future, isolation from acoustic noise, optical feedback, and temperature drifts should lead to oscillator linewidths that approach the Shawlow/Townes limit. An improved experiment is now in progress to measure the oscillator linewidth and frequency stability.

The short 5 mm Nd:YAG rod oscillators have a 16.5 GHz axial mode spacing and yield up to 2 mW of single axial mode output before oscillating in a second axial mode. The temperature tuning rate of the oscillator is 3.1 GHz/ °C. Oscillators as short as 2 mm corresponding to a 41.25 GHz axial mode spacing have been operated. Other materials such as Nd:YLF, Nd:GGG and Nd:Glass may provide better temperature stability than Nd:YAG. However, Nd:YAG is readily available in high optical quality crystals. It has a high gain cross-section that is useful for amplifier design. Nd:YAG also has a 300 GHz gain bandwidth that provides more than adequate tuning range for offset velocity measurements or for FM modulation bandwidth for cw-FM Radar.

In recent experiments a $LiNbO_3$ crystal has been inserted into the Nd:YAG resonator. The electro-optic crystal can be used for harmonic generation or for direct FM modulation of the laser frequency through the electro-optic effect.

A coherent FM Radar system also needs a power oscillator to feed the amplifier stages. A unique out-of-plane ring oscillator was invented[6] to solve the problem of providing high power single axial mode output required of a power amplifier.

The most critical component of a 1 μm coherent Radar system that remained to be demonstrated was a high gain pre-amplifier. Previous approaches to laser amplification had led to a series of amplifier states with each stage isolated from others to prevent self-oscillation. The linear series of independent amplifiers was both complex and inefficient.

One elegant solution to the problem was to use the angular multi-plexing properties of the slab geometry configuration laser medium.[7] A Nd:YAG slab geometry amplifier was designed and tested. Flashlamp pumping at 20 J input energy yielded 23 dB of small signal gain per pass. The expected overall pre-amplifier gain is 60 dB for three passes including losses from beam steering elements. Thus a 10 mW power oscillator can be amplified to the 10 kW power level in a selected 3 μsec pulse duration for 30 mJ of pulse energy. Power amplification in saturated amplifier states could then be used to boost the energy to the level required for coherent radar measurements over an extended range.

The remaining coherent Radar system components have been developed by the rapidly advancing technologies outside of our laboratory. For example, fiber couplers, Faraday isolators, single mode fibers and wide bandwidth detectors are now available.

The Stanford Coherent LIDAR System

Figure 1 shows a schematic of the 1.06 μm Stanford coherent LIDAR system that is now being assembled. The components described above are combined using fiber optical techniques to form a pulsed FM coherent Radar system. The system has all of the components of the classical cw FM Radar systems designed for the microwave wavelength region. The principal difference is the size of the components and the spatial resolution per velocity resolution of the measurement. Both systems size and

the spatial resolution scale with wavelength. Thus the system shown schematically in Fig. 1 is a very compact FM coherent Radar system at one micrometer wavelength.

The system is expected to yield wind measurements in 1 km depth resolved increments to an altitude of 20 km for a transmitted power of 10 kW in 3 μsec long pulses. For these measurements the backscatter coefficient is expected to approach the molecular value of $3 \times 10^{-8}\ m^{-1}$. Wind measurements from an 800 km orbiting platform with a 1 meter diameter receiving aperture require 3 J of transmitted energy for a signal-to-noise of unity.

CONCLUSION

For Radar measurements from hard targets the range can be resolved by providing short transmitted pulses or by FM chirp moldulation of the laser source. The range resolution in either case is ultimately limited by the bandwidth of the laser medium. For Nd:YAG the 300 GHz bandwidth corresponds to a pulse width of 3 psec or to a range resolution of 0.45 mm. This range resolution in combination with the diffraction limited pointing accuracy of a 1 micrometer wavelength yields very accurate target locating capability compared to longer wavelength systems.

Finally, if future Nd:YAG oscillators achieve frequency stabilities of less than 1 kHz, then target vibrometry becomes a practical aspect of the measurement. Prospects for achieving linewidths near the shot noise limit appear good in the near future.

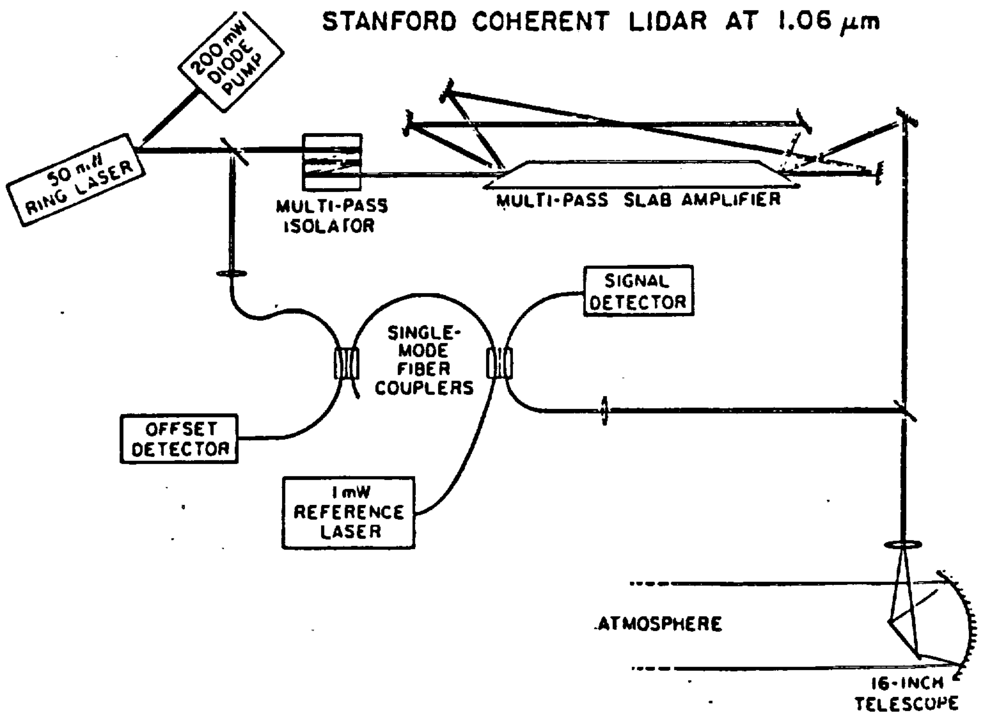

Figure 1. The coherent LIDAR anemometer is shown schematically.

REFERENCES

1. M.I. Skolnik, "Introduction to Radar Systems", McGraw-Hill, New York, 1982.

2. T.J. Kane, B.K. Zhou and R.L. Byer, "Potential for Coherent Doppler Wind Velocity Lidar Using Neodymium Lasers", Applied Optics, 23, p.2477 (1984).

3. R.M. Huffaker, Ed. "Feasibility Study of Satellite Borne Lidar Global Wind Monitoring System", N.O.A.A. Tech. Mem., ERL-WPL-37, (1978).

4. Y.L. Sun and R.L. Byer, "Sub-megahertz Frequency Stabilized Nd:YAG Oscillator", Optics Letters, 7, p.408 (1982).

5. B.K. Zhou, T.J. Kane, G.J. Dixon and R.L. Byer, "Efficient, Frequency Stable Laser Diode Pumped Nd:YAG Laser", Optics Letters, 10, p.62 (1985).

6. T.J. Kane and R.L. Byer, "Monolithic, Unidirectional Single Mode Nd:YAG Ring Laser", Optics Letters, 10, p.65 (1985).

7. T.J. Kane, W.J. Kozlovsky and R.L. Byer, "High Gain Multi-pass Amplification in a Single Nd:YAG Slab", to be presented at the C.L.E.O. Conference, Baltimore, Maryland, May 1985.

The extension of coherent Radar to higher frequencies has been a continuing trend in Radar technology for the past forty years. The recent progress in Nd:YAG based coherent sources is but one more step in the effort to improve the resolution of Radar by extending the techniques of Radar into the optical frequency range.

ACKNOWLEDGEMENTS

This work was supported in part by N.A.S.A. under contract #956757 and in part by Langley under contract #NAG-1-182.

A PROSPECTUS OF SEMICONDUCTOR AND ADVANCED SOLID STATE LASER TECHNOLOGY FOR TRACKING/RENDEZVOUS AND PROXIMITY OPERATIONS

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Schwartz Electro-Optics, Inc., Orlando, FL 32810

This talk will review the latest in the technology of solid state and semiconductor lasers appropriate to the NASA interest in spacecraft tracking/rendezvous and proximity operations. The emphasis will be on devices, not complete systems, with the hope that systems engineers will benefit from an understanding of the newest in device capabilities. Semiconductor lasers are included for two reasons. First, they may be used in arrays as optical pumping sources for the solid state laser, allowing construction of the highly reliable, low-power-consumption lasers needed in a space environment. Second, the semiconductor lasers may themselves be useful in proximity operations since even single diode lasers would have enough output to do accurate ranging at close distances.

The use of solid state lasers, particularly the 1.06-μm wavelength system Nd:YAG, for ranging applications is well established. Typically the lasers are pumped by gas-discharge flashlamps and Q-switched to generate short pulses around 10-20 ns in duration, at repetition rates of up to 100 Hz. Outputs from a single oscillator range up to 200 mJ, and amplifiers can be used to increase the output energy to around 1 J. Nearly diffraction-limited output-beam quality can be obtained if the oscillator uses unstable-resonator optics. Through use of mode-locking or cavity-dumping techniques the Nd laser can produce shorter pulses, in the range 20 ps-1 ns for more accurate ranging data and, by cw pumping and modelocking high pulse rates of ~ 100 MHz can be produced where rapid updating of short ranges is required.

*This work was supported by the Department of the Air Force.

Two developments are of importance in future Nd laser technology. One involves the use of Cr^{3+} ions added to the laser crystal. A series of measurements performed by a group at the Lebedev Institute on various host crystals for the Nd^{3+} ion indicated that Cr^{3+}-sensitized $Gd_3Sc_2Ga_3O_{12}$ (GSGG) is several times more efficient under flashlamp-pumped conditions than the more common host, YAG. However, the data showed a slope (differential power) efficiency for the Nd:Cr:GSGG laser of 2.5%, which can be achieved by a well-designed Nd:YAG laser. To determine if high absolute efficiencies can be obtained from lasers using the sensitized garnet crystal, several experiments were carried out at Lincoln Laboratory employing a 5-mm diameter by 75-mm long laser rod in an efficient optical pumping cavity. The doping level was approximately 1% for both the Nd^{3+} and Cr^{3+} ions. The pump source, a 400-Torr xenon flashlamp with a 4-mm bore and 6.25-inch arc length was driven by a resona tly charged LC discharge circuit with a 55-μF capacitance and 82-μH inductance. The water-cooled pump cavity was constructed of a samarium-doped glass tube with a diffuse reflector coated on the outer surface of the tube.

Slide 1 presents the input-output energy curves for the system running at a repetition rate of 20 Hz for two different flashlamp-wall materials. The laser cavity for these results consisted of two flat mirrors spaced 33 cm apart with an output coupling of 50%; the threshold was 2.6 J. When the output coupling was reduced to 20%, the threshold dropped to ~ 1.2 J, indicating that the crystal losses were relatively low. The laser performance as a function of repetition rate is shown in Slide 2. The rolloff at the highest rate is due to the limited recharging rate of the power supply. Up to 45 W of average power at a 50 Hz rate was observed at ~ 20 J/pulse input energy. Higher outputs could not be obtained because of the limited thermal rating of the pump cavity.

The slope and overall efficiencies of a standard Nd:YAG laser rod in the same pump cavity and with the same pump lamp are 2.5 and 2%, respectively. Thus, the Nd:Cr:GSGG laser, with slope and overall efficiencies of 5.5 and 5%, respectively, was significantly more efficient than YAG when both lasers operated under more optimal conditions than previously reported. Adjustments to the pump-lamp parameters and the rod doping levels may yield even higher efficiencies from the Nd:Cr:GSGG laser.

The other major development in Nd laser technology involves the use of semiconductor lasers as pumping sources. This concept is not new, and was first demonstrated nearly 15 years ago. Recent advances in the technology of semiconductor lasers, specifically the use of quantum-well structures produced by MOCVD systems, have significantly improved the performance of both single-junction lasers and of linear arrays of lasers. The following is based on a paper presented by D. R. Scifres et al. at the CLEO '83 Conference. Slide 3 shows the structure of a 40-emitter array, multiple-quantum well GaAlAs laser. Operation was such that the individual emitters were phase-locked to each other. Slide 4 shows the total power output from one side of the array as a function of total drive current, indicating that 2.5 W of cw output was obtained. For pumping of the Nd laser to reasonable energy output levels arrays will have to be extended to considerably more emitters, and techniques for stacking the linear arrays to form a two-dimensional source will be required. It will also be necessary to set the semiconductor laser wavelength resonant with the absorption bands of the Nd ion. Slide 5 shows the transmission spectrum of a 6-mm-thick sample of Nd:YAG, indicating the presence of numerous absorption lines between 730 and 830 nm suitable for diode pumping. Also shown in the slide is the emission spectrum of a more common pump source for Nd:YAG, the krypton arc lamp.

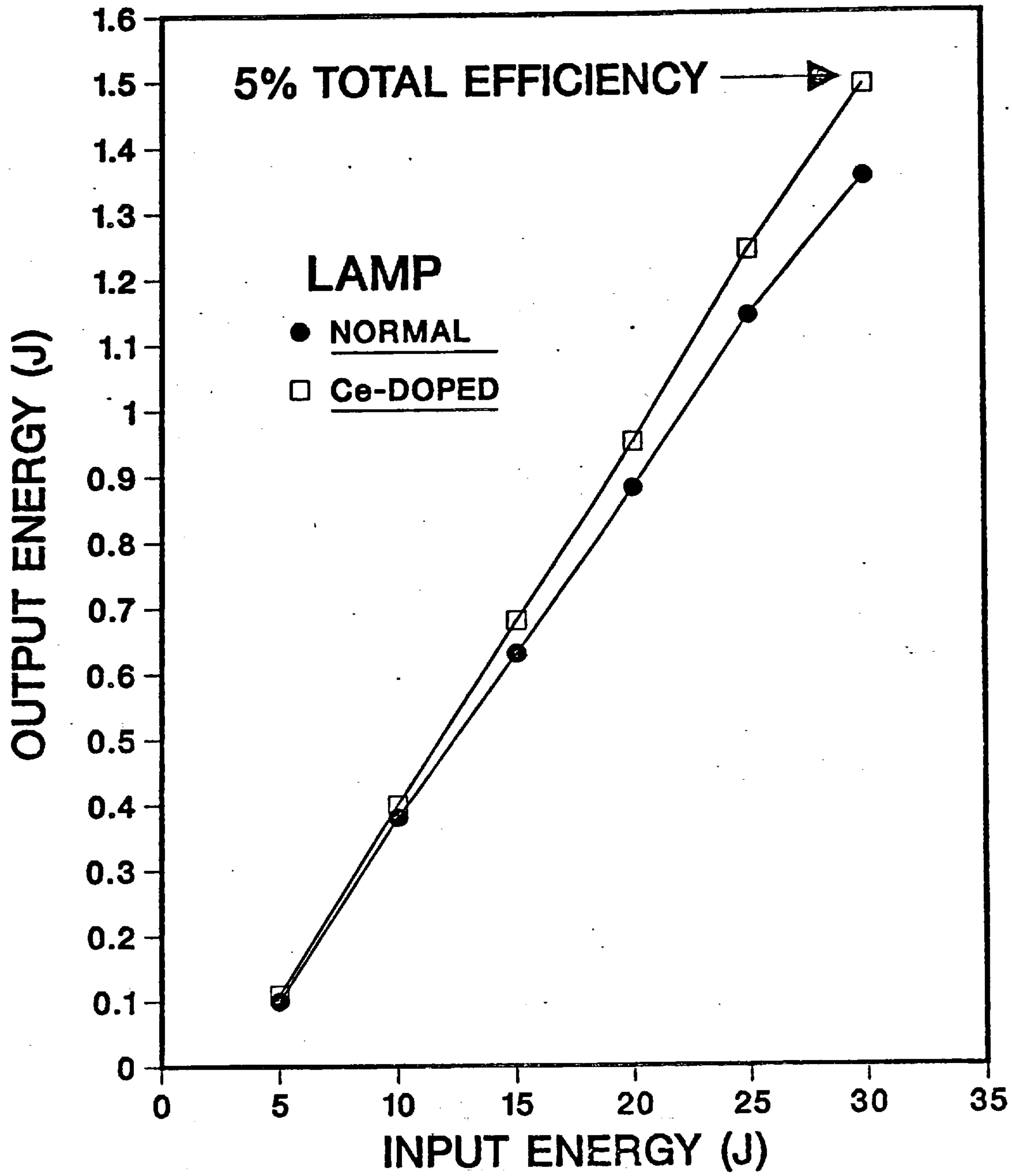
At close ranges, with "closeness" determined by whether the vehicle to be docked has retroreflectors or not, single diode lasers can be used for obtaining range information. When the range measurement is made against a "cooperative" target, that is where the target has a retroreflector, then the diode laser is extremely well suited for range measurements. The use of a retroreflector, changes the return energy from being proportional to the product of target reflectivity and $\frac{1}{R^4}$ to being proportional to $\frac{1}{4R^2}$. The diode laser ranging system has the advantages of being small, simple, inexpensive, reliable, lightweight and highly efficient, requires no high voltages and has long lifetimes.

Laser ranging is usually done on a time-of-flight basis (as with other radar systems), which implies that the speed of light is used to measure the distance between source/detector and the target. This would imply that if one wanted to have 1 cm range resolution then the time resolution would be required to be approximately 67 x 10^{-12} sec. The time-of-flight rangers are generally divided into two categories, pulse and CW systems. Both type of systems can easily yield better than 1 cm accuracy. A pulse laser ranger was reported by I. Kaisto to have a range resolution of $\pm$ 1 mm. The CW laser ranger is widely used in surveying instruments which have absolute accuracy and resolution of a few mm´s and measure ranges out to several kilometers. It

is generally thought that the CW system will yield the best range resolution and accuracy. Surveying instruments use a technique called "time expansion". This is where the return signal is heterodyned with a signal which is slightly offset and phased-locked to the transmitted signal. The net result is that the phase angle which contains the range data is preserved through the heterodyning process and can be measured on the low offset frequency. This technique has now been used for years and can yield time resolution of 10^{-12} seconds. The examples cited above both used device technology developed more than a decade ago.

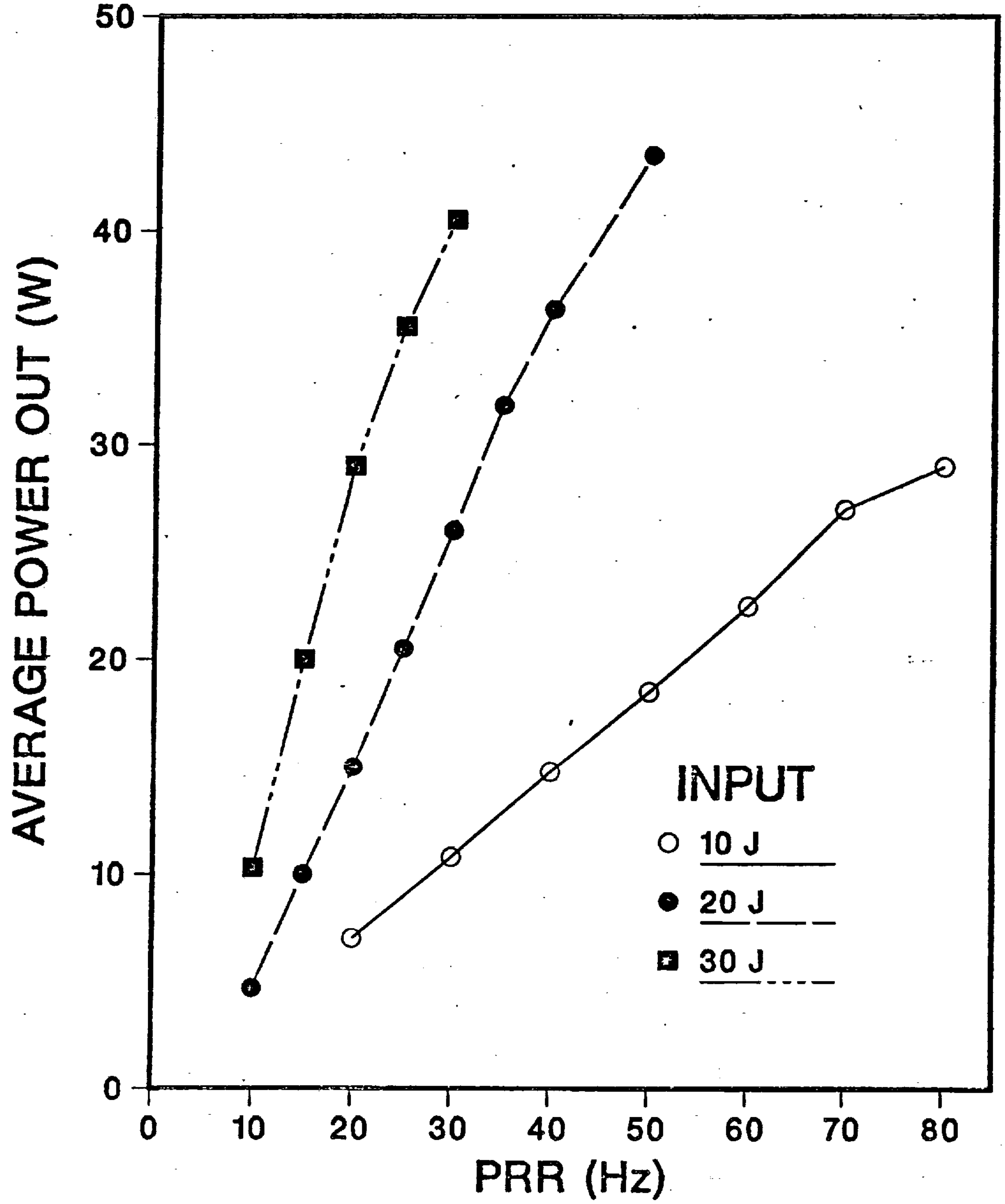
Recently, techniques have been developed for very-high-speed modulation of semiconductor lasers. A recently published paper by D. Z. Tsang et al from Lincoln Laboratory discussed the modulation of laser output up to a frequency of 10.5 GHz, accomplished by an intracavity electroabsorption modulator; Slide 6 diagrams the structure of the modulated laser.

Nd:Cr:GSGG LASER OUTPUT VS INPUT

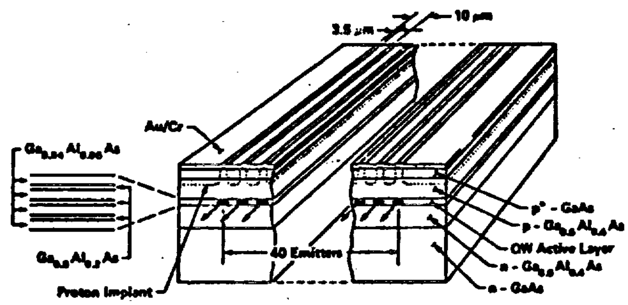


Slide 1

PERFORMANCE OF Nd:Cr:GSGG LASER

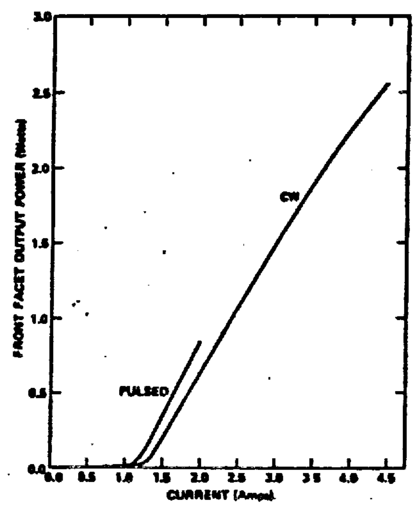


Slide 2

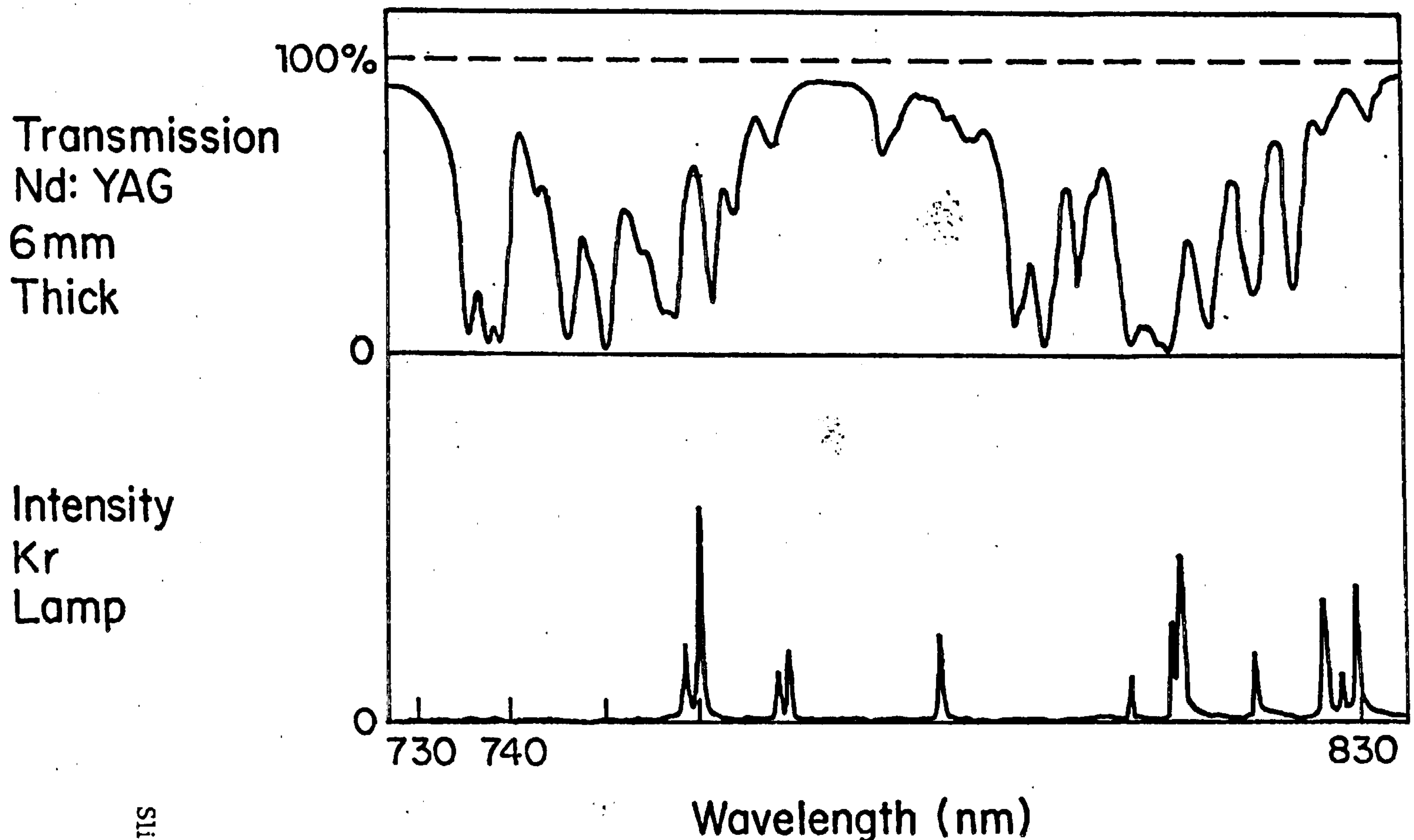


10 µm
3.5 µm
Au/Cr
$Ga_{0.94}Al_{0.06}As$
$Ga_{0.8}Al_{0.2}As$
Proton Implant
40 Emitters
p^{+} - GaAs
p - $Ga_{0.6}Al_{0.4}As$
QW Active Layer
n - $Ga_{0.6}Al_{0.4}As$
n - GaAs

Slide 3

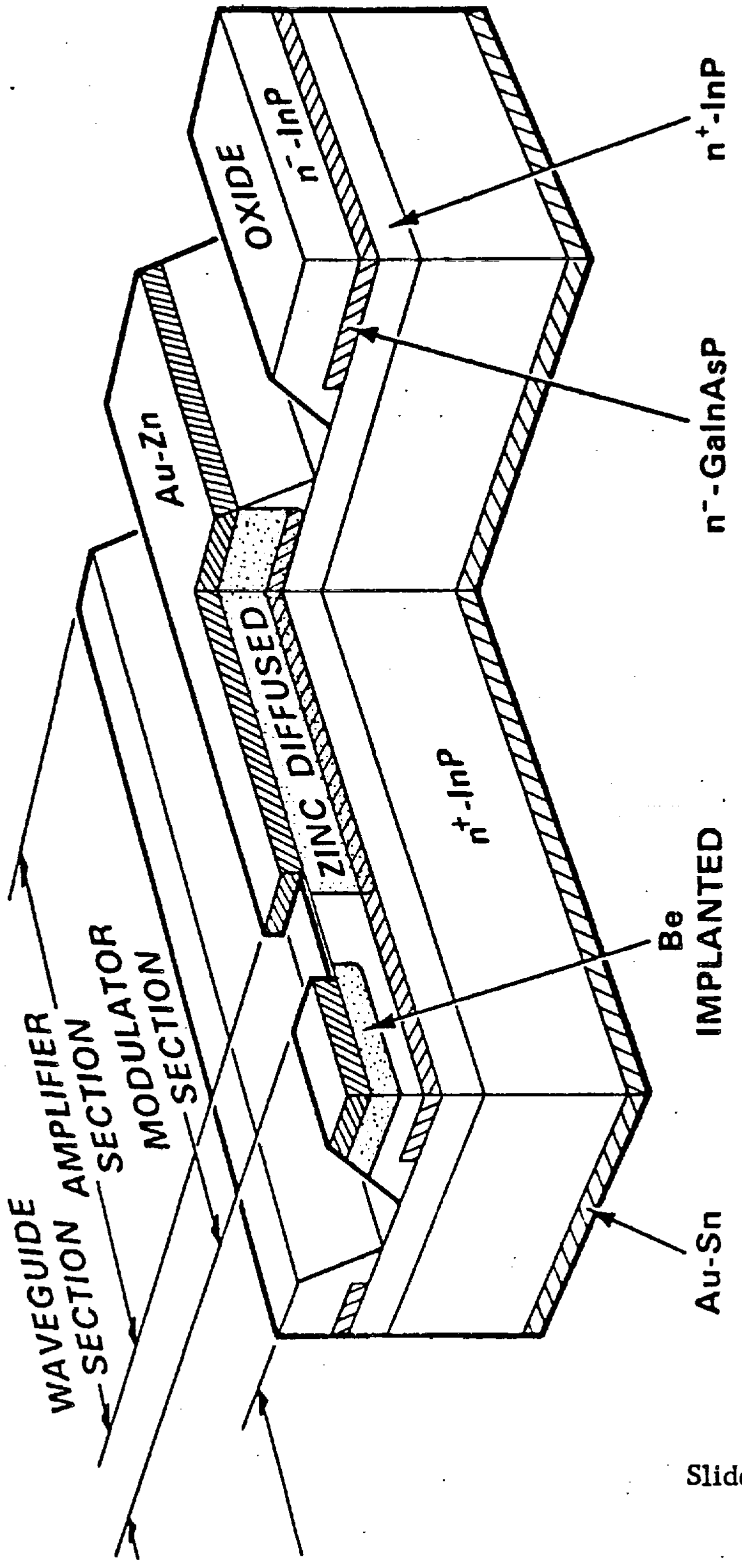

FRONT FACET OUTPUT POWER (Watts)
CURRENT (Amps)
PULSED
CW
0.0
0.5
1.0
1.5
2.0
2.5
3.0
3.5
4.0
4.5

Slide 4



YAG ABSORPTION
Kr ARC EMISSION

Slide 5


WAVEGUIDE SECTION
AMPLIFIER SECTION
MODULATOR SECTION
OXIDE
n^--InP
Au-Zn
ZINC DIFFUSED
n^+-InP
Au-Sn
Be IMPLANTED
n^--GaInAsP
n^+-InP

Slide 6

Long Lifetime Stable CO_2 Lasers for Lidar and Comparison with Nd:YAG Lidar

R. V. Hess, D. R. Schryer, B. D. Sidney, I. M. Miller, G. M. Wood*
B. T. Upchurch**, K. G. Brown***

Summary/Conclusions/Recommendations

Development of long life CO_2 and Nd:YAG lasers over a wide pulse energy range for rendezvous and proximity operation benefits from ongoing programs in satellite based lidar remote sensing for wind velocities aerosols/trace species and geophysical precision ranging. These programs use up to 10 Joule/pulse lasers as prime source, however, low energy pulsed and low power CW lasers are used for heterodyne detection and for control of bandwidth and freqency stability. Pulsed closed cycle 0.67 Joule/pulse, 10 p.p.s. (6.7W average power) CO_2 laser operation with externally heated Pt/SnO_2 catalysts has been demonstrated at 95% steady state power; and low pulse energies ≈ 0.03 Joule/pulse up to 100 p.p.s. lasers have operated for $>10^6$ pulses without deterioration of laser medium heated Pt/SnO_2 catalyst. Lifetimes of 10^4 hours for sealed RF excited 2 to 3 watts CO_2 lasers have been demonstrated at ≈ 70% steady state power. Low CW power/ pulse energy diode laser pumped Nd:YAG lasers and CO_2 lasers for rendezvous and proximity operation have great potential for ≈ 10% efficient ≈ 10^4 hour operation. Low power/pulse energy CO_2 lasers have demonstrated short and long term frequency stability/bandwidth for Doppler lidar measurements of velocities < 1cm/sec; diode laser pumped Nd:YAG laser give short term stability and have great promise for long term stability. Mode locked Nd:YAG lasers operate with shorter pulses than CO_2 lasers and higher range lidar accuracies. Nd:YAG lidar is more compact than CO_2 lidar but has more severe eye safety problems. For higher pulse energies for distant rendezvous/noncooperative targets CO_2 lidar is presently superior, but further research is needed in high power diode laser arrays for Nd:YAG laser pumping.

Low pulse energy/low power CO_2 and diode laser pumped Nd:YAG lasers are ready to permit initiation in FY 86 of comparative studies for range and Doppler lidar in a laboratory test bed, which could also incorporate diode laser lidar. Flight demonstration could be initiated in FY 89. The progress of high pulse energy CO_2 and Nd:YAG lidar development for distant rendezvous is related to progress in lidar for satellite based atmospheric remote sensing.

*NASA/LaRC   **Chemicon Inc.   ***Old Dominion University

6-88

LONG LIFETIME STABLE CO_2 LASERS FOR LIDAR AND COMPARISON WITH Nd:YAG LIDAR

R. V. Hess, D. R. Schryer, B. D. Sidney, I. M. Miller, G. M. Wood,*
B. T. Upchurch**, K. G. Brown***

COMMON TECHNOLOGY WITH LIDAR REMOTE SENSING

- o SATELLITE BASED LIDAR REMOTE SENSING OF WIND VELOCITIES, AEROSOLS, TRACE SPECIES AND GEOPHYSICAL PRECISION RANGING REQUIRES LONG LIFE EFFICIENT CO_2, Nd:YAG AND TUNABLE SOLID STATE LASERS WITH PULSE ENERGIES FROM ABOUT 1/10 TO 10 JOULE AND 10 P.P.S.

- o BUT LOW PULSE ENERGY OR CW POWER LASERS ARE NEEDED AS LOCAL OSCILLATORS FOR HETERODYNE DETECTION AND FREQUENCY/BANDWIDTH CONTROL OF HIGH PULSE ENERGIES LASERS
  - EXAMPLES OF SUCH CONTROLS ARE INJECTION LOCKING OF RESONATOR AND INJECTION OF MASTER OSCILLATOR RADIATION INTO POWER AMPLIFIERS (HESS 1983).

- o THE TECHNOLOGY DEVELOPMENT OF LOW PULSE ENERGY OR CW POWER LONG LIFE EFFICIENT LASERS IS CRUCIAL FOR RENDEZVOUS AND PROXIMITY OPERATION
  - HIGHER PULSE ENERGIES MAY BE REQUIRED FOR DISTANT RENDEZVOUS/NON-COOPERATIVE TARGET MISSIONS.

*NASA/LaRC **CHEMICON, INC. ***OLD DOMINI(UNIVERSITY

LONG LIFETIME STABLE CO_2 LASERS FOR LIDAR AND COMPARISON WITH Nd:YAG LIDAR

LONG LIFE CO_2 LASERS-PULSED

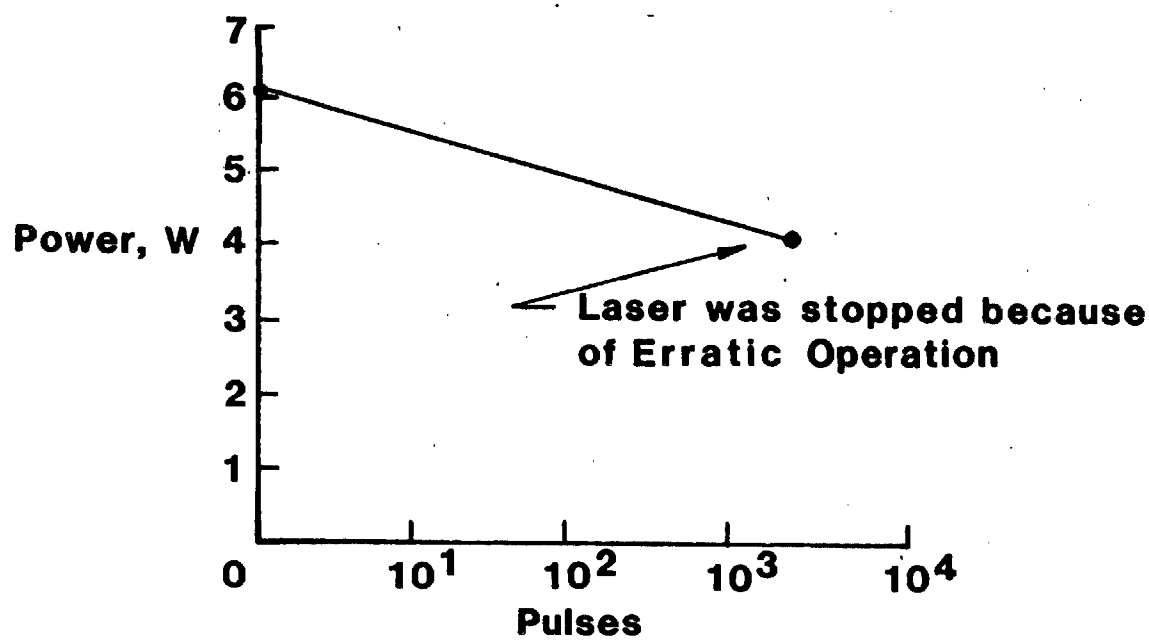
SAMPLE ACCOMPLISHMENTS

- 0.67 JOULE/PULSE, 10 P.P.S. CLOSED CYCLE ATMOSPHERIC CO_2 LASER OPERATION USING 100° TO 200°C Pt/SnO_2 CATALYST WITH 90 TO 95% STEADY STATE POWER; TESTS STOPPED AFTER ≈ 5 HOURS/ 180,000 PULSES, SINCE POWER DID NOT CHANGE (ROGOWSKI 1983) IMPLIES NO CATALYST DETERIORATION.
  - GAS ANALYSIS INDICATES STEADY STATE DUE TO CO/O_2 RECOMBINATION.
- INITIATED RARE ISOTOPE $C^{18}O/^{18}O_2$ RECOMBINATION STUDIES FOR DOPPLER LIDAR REMOTE SENSING WITH CATALYSTS FOR MINIMAL ISOTOPE EXCHANGE (LaRC 1984).
- 0.03 JOULE/PULSE, 10 TO 100 P.P.S. CLOSED CYCLE CO_2 LASER OPERATION FOR Pt/SnO_2 CATALYST HEATED BY CO_2 LASER MIXTURE FOR > 10^6 PULSES, WITHOUT CATALYST DETERIORATION (STARK 1983); NO INFORMATION GIVEN ON STEADY STATE POWER.
- CLAIMS FOR LOW PULSE ENERGY LASERS WITH COMMERCIAL CATALYSTS UP TO 10^7 PULSES

RESEARCH RECOMMENDATIONS

- CLOSED CYCLE PULSED COMMON/RARE ISOTOPE CO_2 LASER OPERATION OVER 10^4 HOURS/3.6 x 10^8 PULSES AT 10 P.P.S., WITH 95% STEADY STATE POWER, USING SOLID OR GAS CATALYSTS AT TEMPERATURES OBTAINED FROM LASER MEDIUM HEATING; AUTOMATED GA AND SURFACE ANALYSIS.

LIFE STUDY OF PULSED HI-ENERGY CO_2 LASER WITHOUT CATALYST



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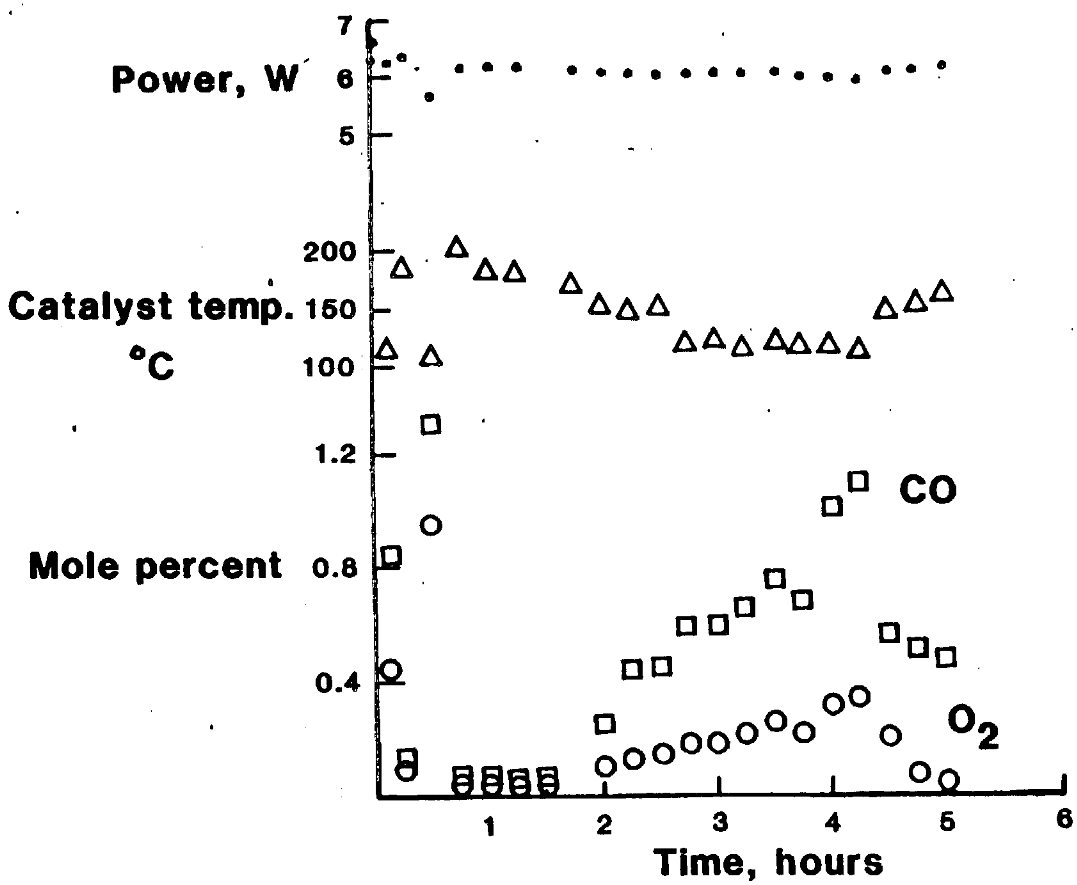


Data for closed cycle operation of CO_2 Laser at 10 pps using platinum on tin oxide catalyst

LONG LIFETIME STABLE CO_2 LASERS FOR LIDAR AND COMPARISON WITH Nd:YAG LIDAR

LONG LIFE CO_2 LASERS - CW

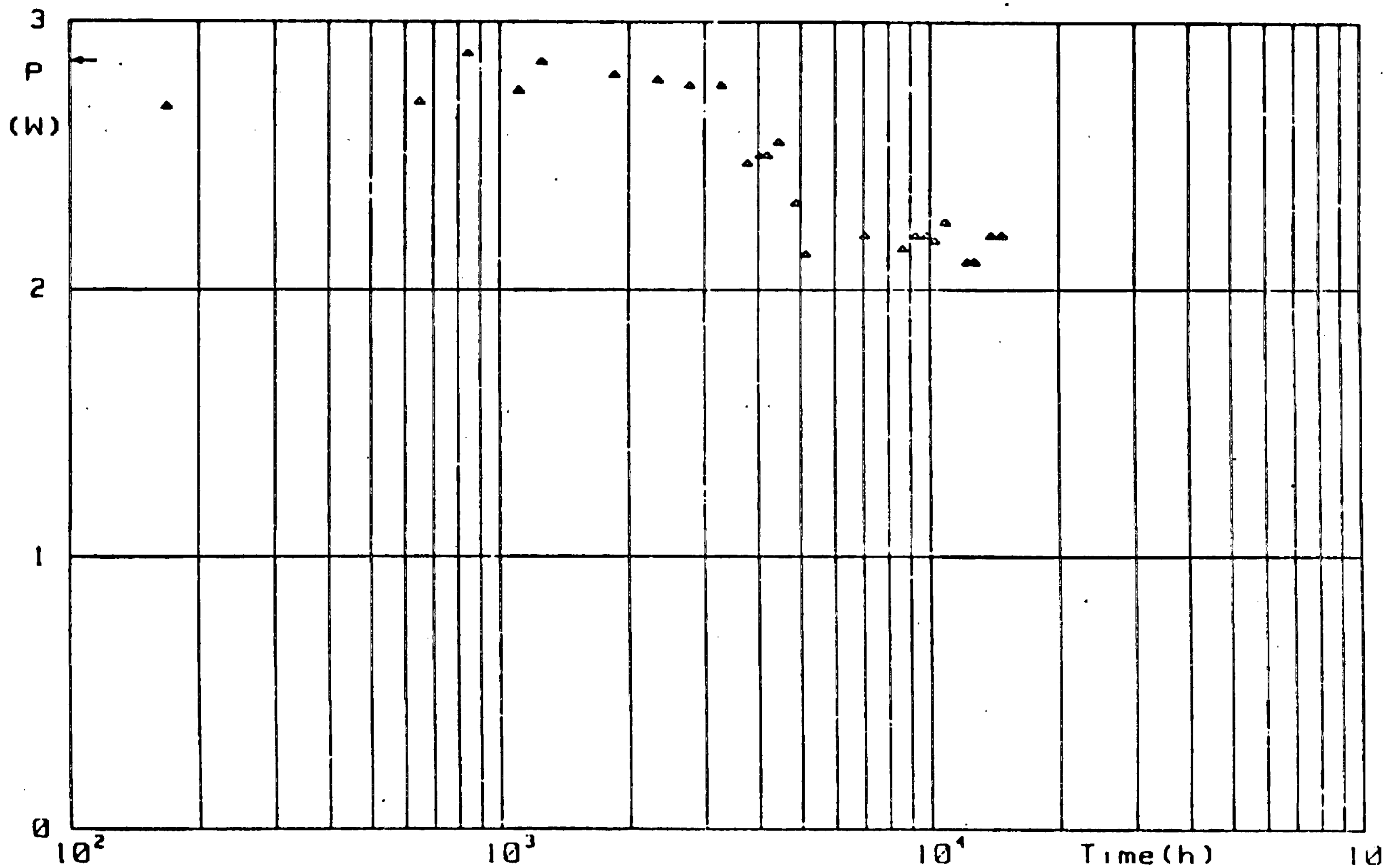
SAMPLE ACCOMPLISHMENTS

- RF EXCITED 2 TO 3W SEALED CO_2 WAVEGUIDE LASER OPERATION FOR ≈ 10^4 HOURS, USING UNSTABLIZED LASERS (HOCHULI 1984)
  - POWER ≈ CONSTANT UP TO ≈ 3×10^3 HOURS, AND ≈ 70% STEADY STATE POWER UP TO ≈ 10^4 HOURS
- DC EXCITED ≈ 1W SEALED CO_2 WAVEGUIDE LASER OPERATION FOR 2.7×10^3 HOURS (LAUGHMAN 1976) AT ≈ 80% STEADY STATE POWER.
- DC EXCITED ≈ 1W SEALED CO_2 WAVEGUIDE LASER OPERATION FOR 10^4 HOURS (HOCHULI 1981) SHOWS INITIAL POWER INCREASE APPARENTLY DUE TO IMPROVEMENT OF LASER MIXTURE (REDUCTION IN H_2O IMPURITIES)

RESEARCH RECOMMENDATIONS

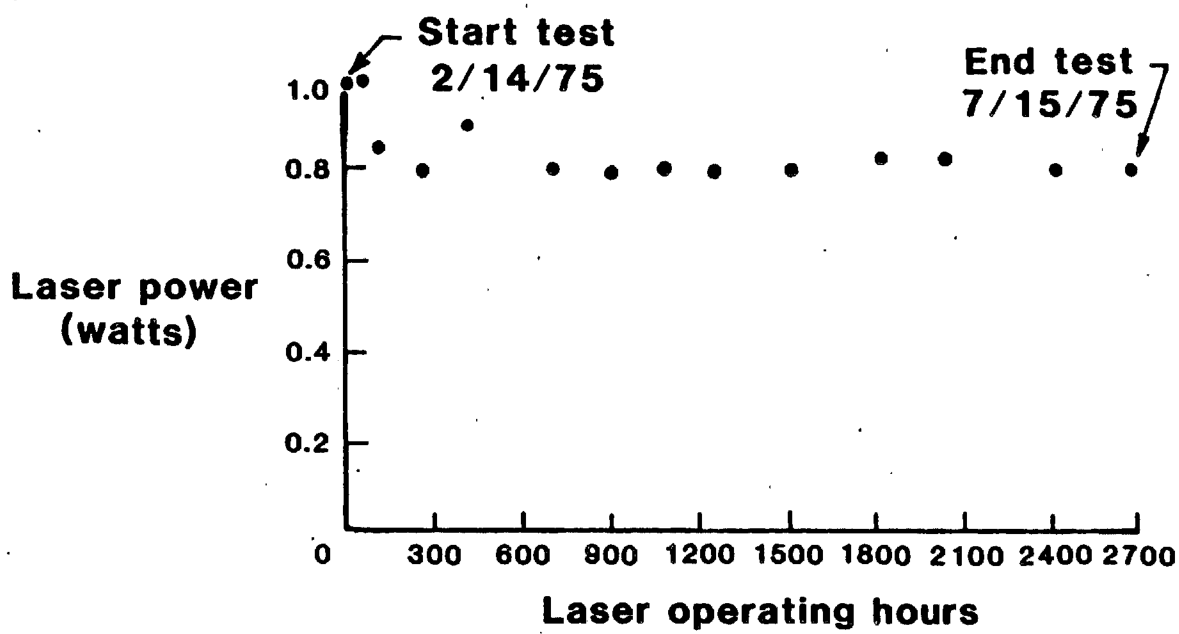
- COMMON AND RARE ISOTOPE CO_2 LASER OPERATION FOR 10^4 HOURS AT SEVERAL WATTS WITH RF AND DC CW EXCITATION AT 95% STEADY STATE POWER, USING SEALED OR CLOSED CYCLE/CATALYST SYSTEM; AUTOMATED GAS AND SURFACE ANAYSIS.
- OPTIMIZE RF EXCITATION EFFICIENCY.

LIFE STUDY OF RF-CW CO_2 WAVEGUIDE LASERS



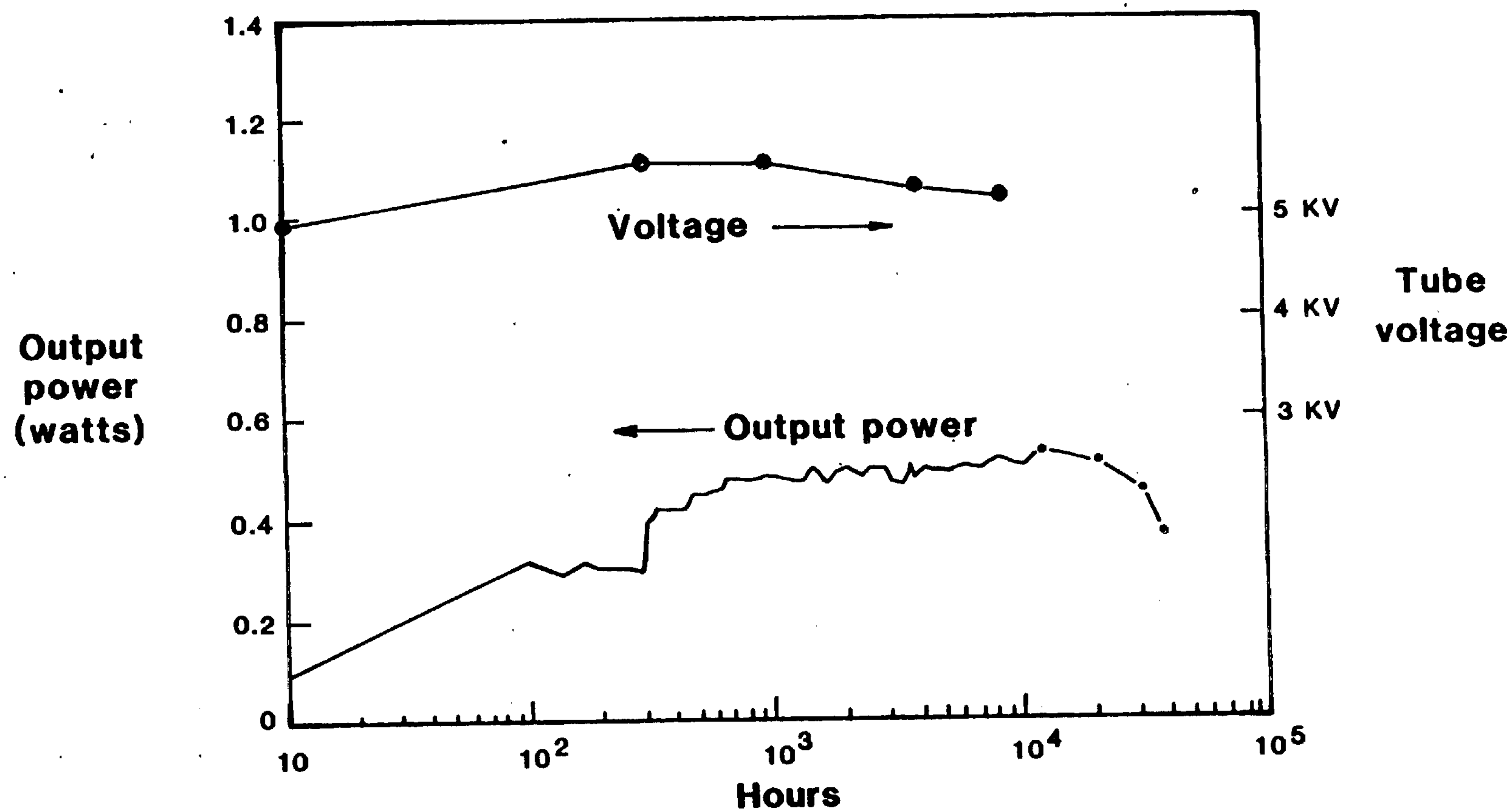
HOCHULI, U.E., 1984

LIFE STUDY OF DC–CW CO_2 WAVEGUIDE LASERS



LAUGHMAN, L.M., 1976

LIFE STUDY OF DC–CW CO_2 WAVEGUIDE LASERS



Laser G2A: Power output and voltage versus time

HOCHULI, V.E., 1981

LONG LIFETIME STABLE CO_2 LASERS FOR LIDAR AND COMPARISON WITH Nd:YAG LIDAR

CO_2 VS Nd:YAG DOPPLER LIDAR

- MEASUREMENTS OF SMALL DOPPLER FREQUENCY SHIFTS $\Delta f_D = 2v/\lambda$ FOR VELOCITIES, v, DOWN TO < 1cm/sec IN RENDEZVOUS AND PROXIMITY OPERATION PUTS STRONG DEMANDS ON SHORT AND LONG TERM FREQUENCY STABILITY/BANDWIDTH FOR DOPPLER LIDAR

- EXTENSIVE APPLICATIONS FOR FREQUENCY/BANDWIDTH CONTROL OF CO_2 LASERS ARE REVIEWED (FREED 1982) AND ALSO DEMONSTRATED BY HELICOPTER BASED DOPPLER NAVIGATION WITH $\approx$ 1cm/sec VELOCITY RESOLUTION (DEL BOCCA 1981)

- DIODE PUMPED Nd:YAG LASERS AT LOW POWERS HAVE DEMONSTRATED REQUIRED FREQUENCY STABILITY/ BANDWIDTH FOR SHORT TIMES (KANE 1984, 1985). LONG TERM ACTIVE STABILIZATION FOR RENDEZVOUS AND PROXIMITY OPERATIONS VERY PROMISING.

- HIGH PULSE ENERGY CO_2 LASERS FOR DOPPLER LIDAR HAVE NEAR TERM POTENTIAL FOR ENGINEERING DEMONSTRATION (HESS 1983; BYRON 1983); DIODE LASER PUMPED Nd:YAG LASERS NEED FURTHER RESEARCH.

LONG LIFETIME STABLE CO_2 LASERS FOR LIDAR AND COMPARISON WITH Nd:YAG LIDAR

CO_2 VS Nd:YAG RANGE LIDAR

- MEASUREMENTS OF HIGH RANGE ACCURACY INTRODUCES SPECIAL REQUIREMENTS FOR HIGH MODULATION FREQUENCIES OR SHORTER PULSES
- TRADE OFFS FOR CO_2 VS Nd:YAG RANGE LIDAR PRESENTED FOR SPECIAL CASE OF MODE LOCKED SHORT PULSES USED IN GEOPHYSICAL PRECISION RANGING Nd:YAG LIDAR (HARPER 1978, 1979)
  - TRADE OFFS FOR DIVERSE MODULATION TECHNIQUES TO BE EVALUATED IN OTHER STUDIES
- MODE LOCKED LASER PULSE DURATION $t \approx 1/\Delta\nu$; $\Delta\nu$ = LASER GAIN BANDWIDTH
  - SINCE $\Delta\nu_{Nd:YAG} > \Delta\nu_{CO_2}$, Nd:YAG LASERS PRODUCE SHORTER PULSES
- MODE LOCKED PULSES PRODUCED BY MODULATION AT CAVITY MODE RESONANCES $\Delta\nu_c = 2L/C$, WHICH ALSO EQUALS PULSE REPETITION FREQUENCY
  - REDUCTION OF EXCESSIVELY HIGH PULSE REPETITION FREQUENCIES OBTAINED BY SELECTION OF SINGLE PULSE UNDER Q-SWITCHED ENVELOPE; SINGLE PULSE ENERGY INCREASED THROUGH MOPA AMPLIFICATION.

LONG LIFETIME STABLE CO_2 LASERS FOR LIDAR AND COMPARISON WITH Nd:YAG LIDAR

CO_2 VS Nd:YAG LIDAR: LIFE, EFFICIENCY, EYE SAFETY

- LOW PULSE ENERGY OR CW POWER CO_2 LIDAR AND DIODE LASER PUMPED Nd:YAG LIDAR HAVE GREAT POTENTIAL FOR 10^4 HOURS LIFE AND EFICIENCIES ≈ 10%
  - HIGH PULSE ENERGY CO_2 LIDAR EFICIENCY ≈ 5%; 10^4 HOUR LIFE CLOSED CYCLE OPERATION REQUIRES NEAR TERM CATALYST RESEARCH AND ENGINEERING DEMONSTRATION.
  - HIGH PULSE ENERGY DIODE LASER PUMPED Nd:YAG LASERS REQUIRE LONG TERM RESEARCH FOR FEASIBILITY DEMONSTRATION.
- Nd:YAG LIDAR CONSIDERABLY MORE COMPACT, BUT MORE SEVERE EYE SAFETY PROBLEMS FOR 1.06μm Nd:YAG THAN FOR 9–11μm CO_2 LIDAR.

RECOMMENDATIONS

- INITIATE COMPARATIVE STUDIES OF LONG LIFE CO_2 AND DIODE LASER PUMPED Nd:YAG RANGE AND DOPPLER LIDAR IN LABORATORY TEST BED FOR RENDEZVOUS AND PROXIMITY OPERATION - FY 86.
  - TEST BED COULD ALSO INCLUDE DIODE LASER LIDAR
- INITIATE FLIGHT DEMONSTRATION - FY 89

REFERENCES

Byron, S. R.; Moody, S. E.; Znotins, T. A.: Definition Study of the WINDSAT Laser Subsystem. 2nd Topical Meeting on Coherent Laser Radar. Technology and Applications, August 1983.

Del Bocca, R. L.; Mongeon, R. J.: Multifunction CO_2 Laser Radar Technology. SPIE Proc., Vol. 300, 19-32, Aug. 1981.

Freed, C.: Frequency Stabilization of CO_2 Lasers; Some Applications. SPIE Proc., Vol. 335, 59-68, 1982.

Harper, L. L.; Williams, R. H.; Logan, K. E.; Stevens, D. A. (ILS Inc.) and Degnan, J. J. (NASA/GSFC): Ultrashort Pulse Solid-State Transmitter Development. Proc. Int. Conf. on Lasers 78, Dec. 1978.

Harper, L. L.; Logan, K. E.; Williams, R. H.; Stevens, D. A.: Nd:YAG Development for Spaceborne Laser Ranging System. NASA-CR-159932/NAS5-22916. Final Report Jan. 1976-Feb. 1979.

Hess, R. V.; Brockman, P.; Bair, C. H.; Staton, L. D.; Lytle, C. D. (NASA/LaRC); Laughman, L. M. (CLS Laser Systems Inc.); Kaplan (Systems and Applied Sciences Corp.): Coherent Infrared Lidar Mission and Technology Needs for Measurements of Transport and Concentration of Tropospheric Trace Species. SPIE Proc., Vol. 415, 52-70, April 1983.

Hochuli, U.: Continued Life Test Results for an Ensemble of CO_2 Lasers. Final Report NASA Grant NSG 5042/Degnan/GSFC, April 1981.

Hochuli, U.: Investigation of RF Excited CW CO_2 Waveguide Lasers. Progress Report NASA Grant NAG5-263/Degnan/GSFC, April 1984.

Kane, T. L.; Zhou, B.; Byer, R. L.: Potential for Coherent Doppler Wind Velocity Lidar Using Neodymium Lasers. Appl. Opt., Vol. 23, No. 11, 2477-2481, 1984.

Kane, T. L.; Byer, R. L.: Doppler Lidar Using Nd:YAG Technology. Topical Meeting on Remote Sensing of the Atmosphere FB3, Lake Tahoe, NV, Jan. 1985.

REFERENCES (continued)

Laughman, L. M.: Long Term Operation of a Sealed Waveguide CO_2 Laser. Rev. Sci. Instrum., Vol. 47, No. 11, 1411-1413, 1976.

Rogowski, R. S.; Miller, I. M.; Wood, G. M.; Schryer, D. R.; Hess, R. V. (NASA/LaRC) and Upchurch, B. T. (Chemicon, Inc.): Evaluation of Catalyst for Closed Cycle Operation of High Energy Pulsed Lasers. SPIE Proc., Vol. 415, 112-117, April 1983.

Stark, D. S.; Crocker, A.; Steward, G. J.: A Sealed 100Hz CO_2 TEA Laser Using High CO_2 Concentrations and Ambient Temperature Catalysts (RSRE). J. Phys. E. Sci. Instrum., Vol. 16, 1983.

LONG LIFETIME STABLE CO_2 LASERS FOR LIDAR AND COMPARISION WITH Nd:YAG LIDAR

R. V. Hess*, D. R. Schryer*, I. M. Miller*, B. D. Sidney*
G. M. Wood*, B. T. Upchurch**, K. G. Brown***

MAIN TOPICS

- COMMON TECHNOLOGY WITH ONGOING LIDAR REMOTE SENSING PROGRAMS
- LONG LIFE CO_2 LASERS - PULSED
- LONG LIFE CO_2 LASERS - CW
- CO_2 VS Nd:YAG DOPPLER LIDAR AND LASER INTERFEROMETRY
- CO_2 VS Nd:YAG AND DIODE RANGE LIDAR
- RECOMMENDATIONS FOR FUTURE STUDIES

*NASA Langley Research Center **Chemicon Inc. ***Old Dominion University

LONG LIFETIME STABLE CO_2 LASERS FOR LIDAR AND
COMPARISON WITH Nd:YAG LIDAR

COMMON TECHNOLOGY WITH ONGOING LIDAR REMOTE SENSING PROGRAMS

- SATELLITE BASED LIDAR REMOTE SENSING OF WIND VELOCITIES, TRACE SPECIES AND GEOPHYSICAL PRECISION RANGING REQUIRES LONG LIFE, EFFICIENT, STABLE CO_2, Nd:YAG AND TUNABLE SOLID STATE LASERS WITH PULSE ENERGIES FROM ABOUT 1/10 TO 10 JOULE AND 10 P.P.S. AND PULSE DURATIONS FROM $\approx 10^{-10}$ TO 10^{-6} secs.

- BUT LOW PULSE ENERGY OR CW POWER LASERS ARE NEEDED AS LOCAL OSCILLATORS FOR HETERODYNE DETECTION AND FREQUENCY/BANDWIDTH CONTROL OF HIGH PULSE ENERGY LASERS
  - EXAMPLES OF SUCH CONTROLS ARE INJECTION LOCKING OF RESONATORS AND INJECTION OF MASTER OSCILLATOR RADIATION INTO POWER AMPLIFIERS (HESS 1983).

- THE DEVELOPMENT/USE OF LOW PULSE ENERGY OR CW POWER LASERS WITH CONTROLLED OUTPUT IS CRUCIAL FOR RENDEZVOUS AND PROXIMITY OPERATION AND ROBOTICS.
  - HIGHER ENERGIES ARE REQUIRED FOR DISTANT RENDEZVOUS/NON-COOPERATIVE TARGETS.

LONG LIFETIME STABLE CO_2 LASERS FOR LIDAR AND COMPARISON WITH Nd:YAG LIDAR

LONG LIFE CO_2 LASERS-PULSED

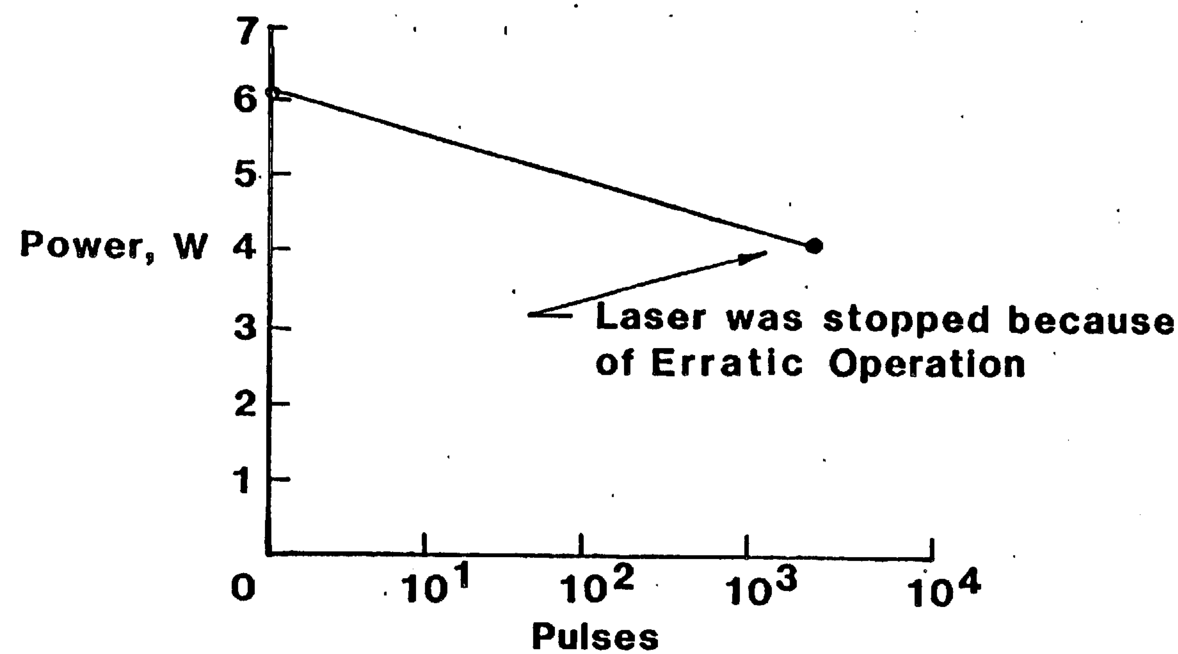
SAMPLE ACCOMPLISHMENTS

- 0.67 JOULE/PULSE, 10 P.P.S. CLOSED CYCLE ATMOSPHERIC CO_2 LASER OPERATION USING 100° TO 200°C Pt/SnO_2 CATALYST WITH 90 TO 95% STEADY STATE POWER; TESTS STOPPED AFTER ≈ 5 HOURS/180,000 PULSES, SINCE POWER DID NOT CHANGE (ROGOWSKI 1983) IMPLIES NO CATALYST DETERIORATION.
  - GAS ANALYSIS SHOWS STEADY STATE CO/O_2 RECOMBINATION; <0.5% O_2 UNIFORM DISCHARGE.
- SAME LASER WITHOUT CATALYST, BUT CO ADDITION AT ≈ 80% STEADY STATE POWER (LaRC 1984).
- INITIATED RARE ISOTOPE $C^{18}O/^{18}O_2$ RECOMBINATION STUDIES FOR DOPPLER LIDAR REMOTE SENSING WITH CATALYSTS FOR MINIMAL ISOTOPE EXCHANGE (LaRC 1984).
- 0.03 JOULE/PULSE, 10 TO 100 P.P.S. CLOSED CYCLE CO_2 LASER OPERATION FOR Pt/SnO_2 CATALYST HEATED BY CO_2 LASER MIXTURE FOR > 10^6 PULSES, WITHOUT CATALYST DETERIORATION (STARK 1983).
- CLAIMS FOR LOW PULSE ENERGY LASERS WITH COMMERCIAL CATALYSTS UP TO 10^7 PULSES

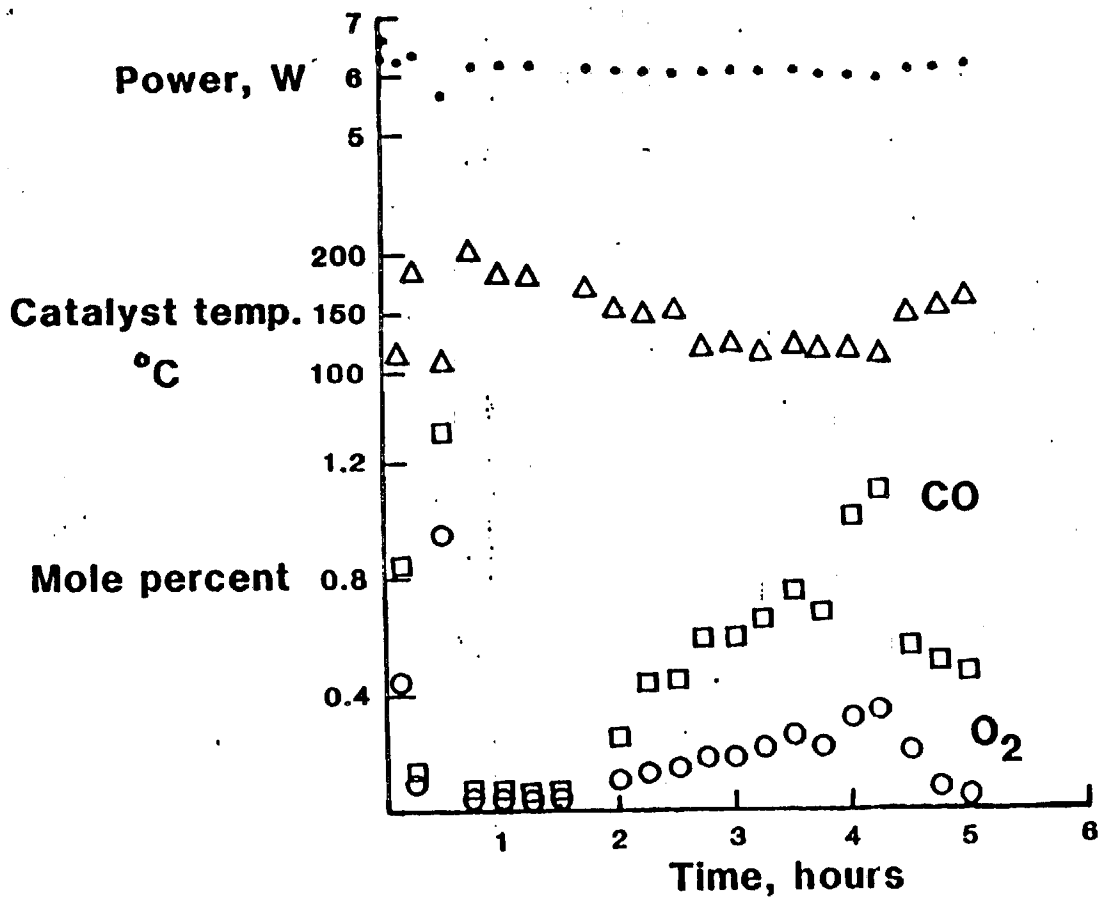
RESEARCH RECOMMENDATIONS

- CLOSED CYCLE PULSED COMMON/RARE ISOTOPE CO_2 LASER OPERATION OVER 10^4 HOURS/3.6×10^8 PULSES AT 10 P.P.S., WITH 95% STEADY STATE POWER, USING SOLID OR GAS CATALYSTS AT LASER TEMPERATURES.

LIFE STUDY OF PULSED HI-ENERGY CO_2 LASER WITHOUT CATALYST



LIFE STUDY OF PULSED HI-ENERGY CO_2 LASER WITH CATALYST



Data for closed cycle operation of CO_2 Laser at 10 pps using platinum or tin oxide catalyst

LONG LIFETIME STABLE CO_2 LASERS FOR LIDAR AND COMPARISON WITH Nd:YAG LIDAR

LONG LIFE CO_2 LASERS - CW

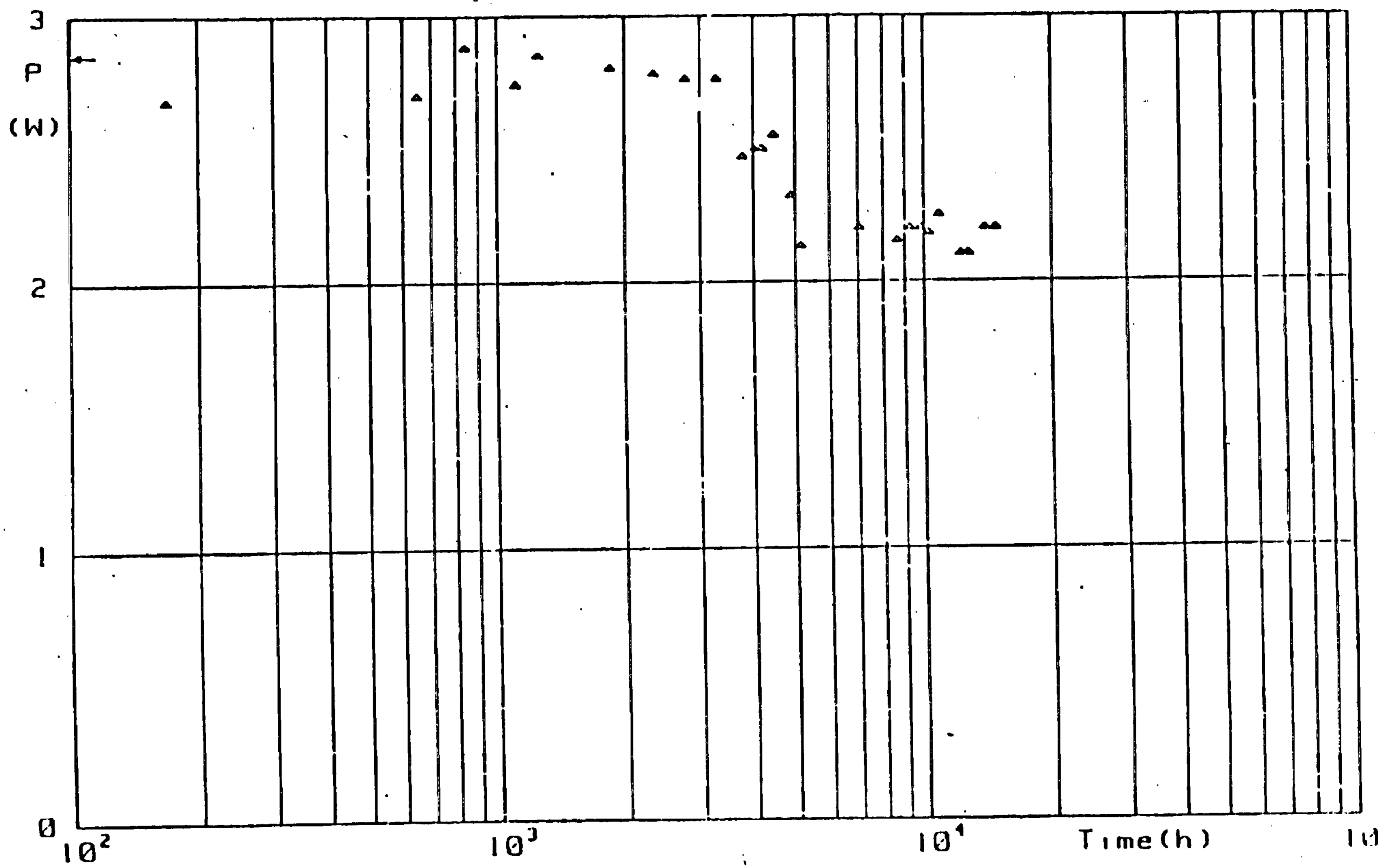
SAMPLE ACCOMPLISHMENTS

- RF EXCITED 2 TO 3W SEALED $\approx$ 120 TORR CO_2 WAVEGUIDE LASER FOR $>10^4$ HOURS, (HOCHULI 1984)
  - STEADY STATE POWER $\approx$ SAME UP TO $\approx 3 \times 10^3$ HOURS, $\approx$ 70% UP TO $\approx 10^4$ HOURS
  - PROPOSED INTEGRATED COMPACT LASER CONSTRUCTION (HOCHULI, 1983)
- DC EXCITED $\approx$ 1W SEALED CO_2 WAVEGUIDE LASER OPERATION FOR 2.7×10^3 HOURS (LAUGHMAN 1976) AT $\approx$ 80% STEADY STATE POWER.
- SOME COMMERCIAL CW LASERS SHOW LONG LIFETIME (INCL. SHELFLIFE), HOWEVER, WITH CONSIDERABLE STEADY STATE POWER REDUCTION

RESEARCH RECOMMENDATIONS

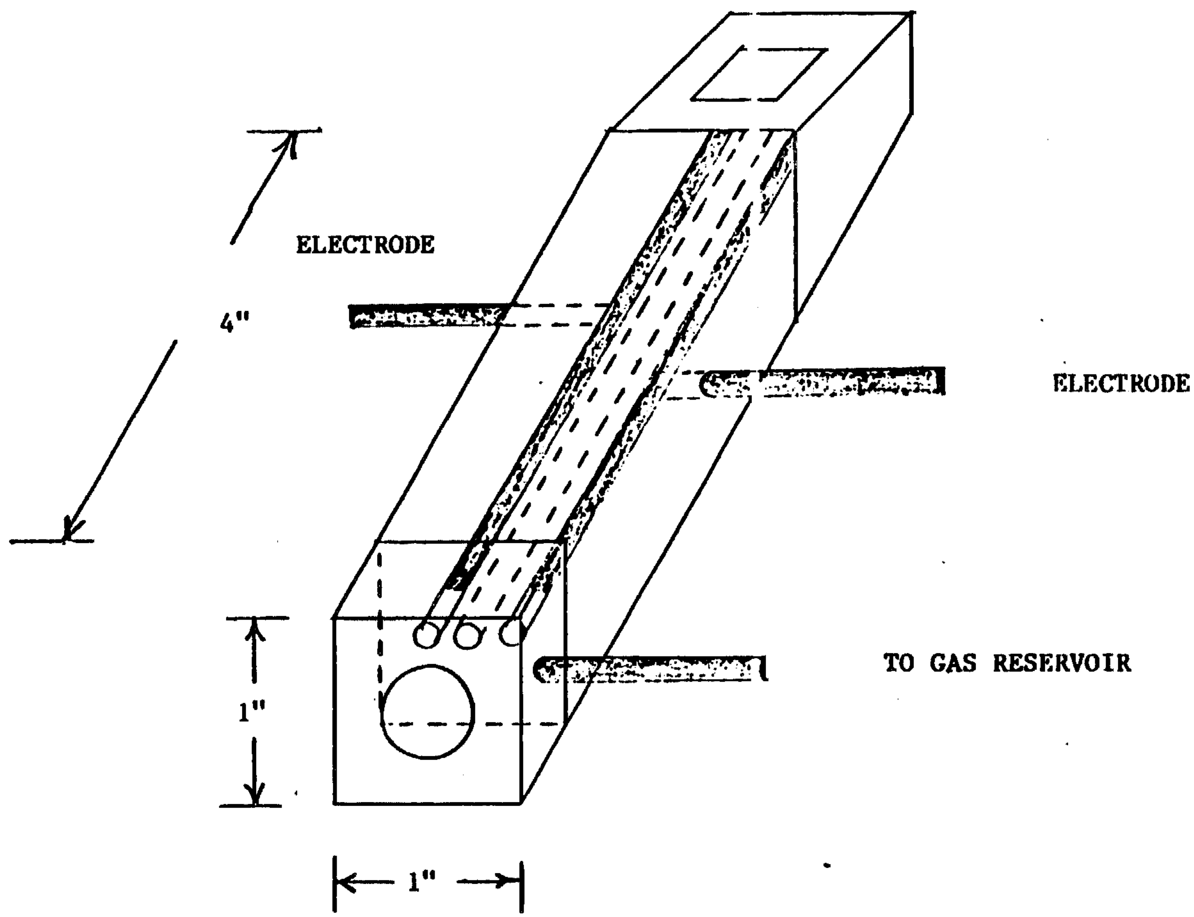
- STRONG NEED OF IMPROVED DIAGNOSTICS (GAS/SURFACE) IN SEALED CW CO_2 LASERS TO EVALUATE REDUCTION OF STEADY STATE POWER DUE TO DISSOCIATION, SPUTTERING, OPTICAL SURFACES USING DIODE LASERS, MASS SPECTROMETER/GAS CHROMATOGRAPH.
- COMMON AND RARE ISOTOPE CO_2 LASER OPERATION FOR $\approx 10^4$ HOURS AT 95% STEADY STATE POWER WITH SEALED OR CLOSED CYCLE/CATALYST SYSTEM; OPTIMIZE RF EXCITATION EFFICIENCY.

LIFE STUDY OF RF-CW CO_2 WAVEGUIDE LASERS



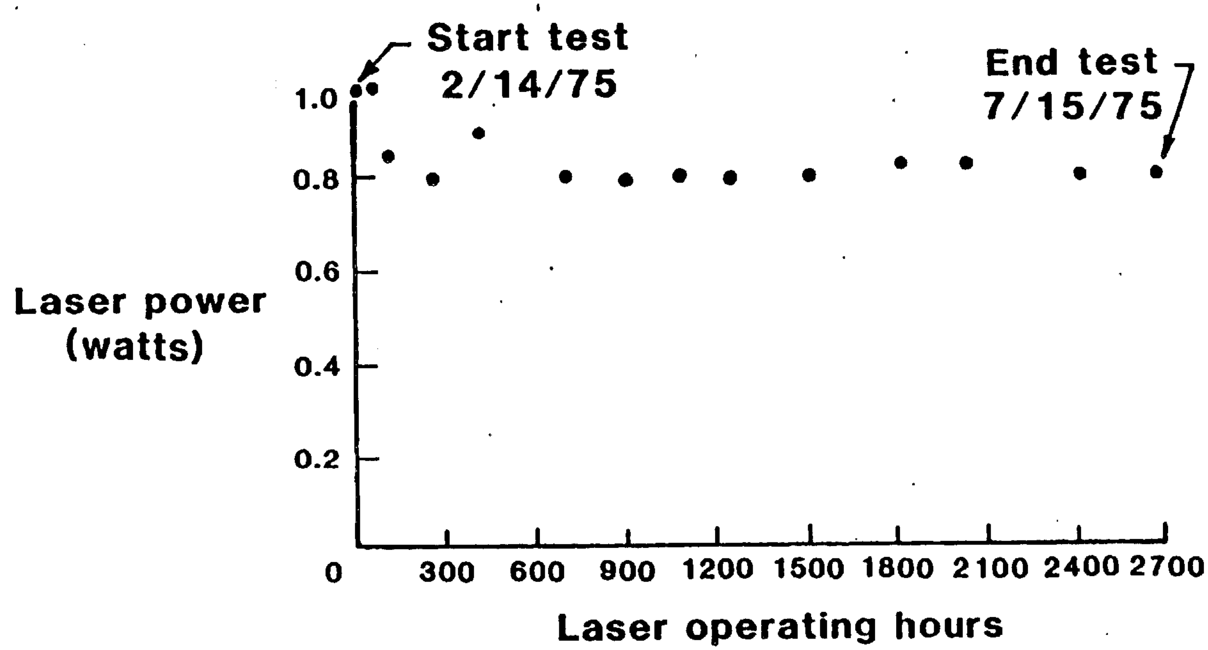
HOCHULI, U.E., 1984

INTEGRATED STRUCTURE



HOCHULI, U. E., 1985

LIFE STUDY OF DC-CW CO_2 WAVEGUIDE LASERS



LAUGHMAN, L.M., 1976

LONG LIFETIME STABLE CO_2 LASERS FOR LIDAR AND COMPARISON WITH Nd:YAG LIDAR

CO_2 VS Nd:YAG DOPPLER LIDAR AND LASER INTERFEROMETRY

- MEASUREMENTS OF SMALL DOPPLER FREQUENCY SHIFTS $\Delta f_D = 2v/\lambda$ FOR VELOCITIES, v, DOWN TO ≈1mm/sec IN PROXIMITY OPERATION PUT STRONG DEMANDS ON FREQUENCY STABILITY AND NARROW BANDWIDTH FOR DOPPLER LIDAR.

- ANALYSIS OF CO_2 LIDAR RETURNS FROM GEOS III SATELLITE AT RANGE OF 1063KM FROM M.I.T. LINCOLN LAB. FIREPOND YIELDED ACCURACY OF ≈0.5mm/sec OF 8KM/sec VELOCITY (FREED 1982)
  - OTHER EXAMPLE OF SHORT TERM FREQUENCY/BANDWIDTH STABILITY DEMONSTRATED BY HELICOPTER BASED DOPPLER NAVIGATION WITH ≈ 1cm/sec VELOCITY ACCURACY (DEL BOCCA 1981)

- IF LONG TERM STABILITY/BANDWIDTH CONTROL NEEDED, LOCK TO NARROW REFERENCE LINE
  - LONG TERM FREQUENCY STABILITY ≈30Hz LASER FLUCTUATION OBTAINED FOR ≈40 secs USING 4.3μm FLUORESCENCE STABILIZATION TECHNIQUE (FREED 1982).

- LASER INTERFEROMETER FOR VERY HIGH ACCURACY MEASUREMENTS REQUIRES LONG COHERENCE LENGTH/NARROW LASER BANDWIDTH.

LONG LIFETIME STABLE CO_2 LASERS FOR LIDAR AND COMPARISON WITH Nd:YAG LIDAR

CO_2 VS Nd:YAG DOPPLER LIDAR

- DIODE LASER PUMPED Nd:YAG LASERS AT LOW POWERS HAVE YIELDED $\Delta f_D \approx$ 200KHz FREQUENCY STABILITY OVER $\approx$ 0.1 sec (KANE 1984), CORRESPONDING TO VELOCITY ACCURACY $v \approx (\Delta f \lambda)/2 \approx$ 100cm/sec
  - RECENT REDUCTION OF FREQUENCY JITTER TO 10KHz FOR 0.3sec (KANE 1985) AND FURTHER ADVANCES IN BYER'S PRESENTATION OFFER GREAT PROMISE FOR CLOSE PROXIMITY OPERATION.
  - LONG TERM FREQUENCY STABILITY CAN BE OBTAINED BY LOCKING FREQUENCY DOUBLED 1.06µm Nd:YAG LASER TO IODINE LINE (LEVENSON, 1972). TECHNIQUE HAS BEEN DEMONSTRATED FOR 0.633µm He-Ne LASER. (TANAKA, 1979).

- LOW POWER DIODE LASER PUMPED Nd:YAG LIDAR GREAT POTENTIAL FOR $>10^4$ HOUR LIFETIME

- HIGH PULSE ENERGY CO_2 LASERS FOR DOPPLER LIDAR POTENTIAL FOR ENGINEERING DEMONSTRATION (HESS 1983, BYRON, 1983). DIODE LASER PUMPED Nd:YAG LASERS NEED FURTHER RESEARCH.

- DIODE LASER PUMPED Nd:YAG LIDAR MORE COMPACT, BUT MORE SEVERE EYE SAFETY FOR 1.06µm Nd:YAG THAN FOR 9-11µm CO_2 LIDAR; IF NEEDED, IMPROVED EYE SAFETY >1.4µm SOLID STATE LASERS.

LONG LIFETIME STABLE CO_2 LASERS FOR LIDAR AND COMPARISON WITH Nd:YAG LIDAR

CO_2 VS Nd:YAG RANGE LIDAR

- o MEASUREMENTS OF HIGH RANGE ACCURACY INTRODUCES SPECIAL REQUIREMENTS FOR HIGH MODULATION FREQUENCIES OR SHORTER PULSES
- o LIMIT OF MODULATION PERIOD T ≈ MODE LOCKED PULSE DURATION ≈ INVERSE GAIN BANDWIDTH $\Delta\nu$ OF LASER
  - SINCE $\Delta\nu_{Nd:YAG} > \Delta\nu_{CO_2}$, T IS SMALLER FOR Nd:YAG (10^{-10}sec) THAN FOR ≈1ATM CO_2 LASERS LASERS (≈ 10^{-9}sec) AND EVEN SMALLER FOR Nd:GLASS (≈10^{-11}sec) AND DIODE (≈10^{-11}sec) LASERS.
  - FOR HIGH SIGNAL TO NOISE RATIOS, RANGE ACCURACY ≈ CT/2 CAN BE FURTHER IMPROVED THROUGH INCREASED ACCURACY OF PHASE OR TIME OF FLIGHT MEASUREMENTS.
  - NEW TECHNIQUES YIELD ≈10^{-12}sec PULSE DURATIONS FOR CO_2 LASERS (CORKUM 1985) AND OTHERS.
- o TO AVOID RANGE AMIBGUITY AT HIGH MODULATION OR PULSE REPETITION FREQUENCIES, SELECT SINGLE MODE LOCKED PULSE UNDER Q-SWITCHED ENVELOPE AND INCREASE ENERGY/SNR THROUGH MASTER OSCILLATOR POWER AMPLIFIER; ALSO REDUCED INTEGRATION TIME; IMPROVED RANGE RATE, MULTI-BEAM OPERATION.
  - TECHNIQUE USED IN GEOPHYSICAL PRECISION RANGING Nd:YAG LIDAR (HARPER/GSFC 1978/79)
  - REDUCTION OF HIGH MODULATION/REPETITION FREQUENCY AT HIGH SNR FOR Q-SWITCHED CO_2, Nd:YAG, Nd:GLASS LASERS OFFERS CERTAIN ADVANTAGES OVER THE SIMPLER DIODE LASER SYSTEMS.

LONG LIFETIME STABLE CO_2 LASERS FOR LIDAR AND COMPARISION WITH Nd:YAG LIDAR

RECOMMENDATIONS FOR FUTURE STUDIES

- INITIATE COMPARATIVE CO_2, DIODE LASER PUMPED Nd:YAG AND DIODE LIDAR STUDIES IN LABORATORY DEMONSTRATIONS OF CRITICAL TECHNOLOGIES FOR RENDEZVOUS AND PROXIMITY OPERATIONS.
- EVALUATE TRADE-OFFS BETWEEN IMPROVED CONTROL OF FREQUENCY STABILITY/BANDWIDTH AND PULSED CW OPERATION, SIGNAL TO NOISE RATIO FOR CO_2 AND DIODE LASER PUMPED Nd:YAG LIDAR VS DIODE LIDAR.
- EVALUATE COMPARATIVE READINESS OF CO_2, Nd:YAG AND DIODE LIDAR FOR RENDEZVOUS AND PROXIMITY OPERATION AND POTENTIAL BENEFITS TO OTHER SPACE STATION APPLICATIONS, SUCH AS ROBOTICS.
- SELECT MOST PROMISING LASER SYSTEM FOR APPLICATION TO LIDAR FOR RENDEZVOUS AND PROXIMITY OPERATIONS AND PERFORM FLIGHT DEMONSTRATIONS.

REFERENCES

Byron, S. R.; Moody, S. E.; Znotins, T. A.: Definition Study of the WINDSAT Laser Subsystem. 2nd Topical Meeting on Coherent Laser Radar. Technology and Applications, August 1983.

Corkum, P. B.: Amplification of Picoseasonal 10μm Pulses in Multiatmosphere CO_2 Lasers. IEEE Jl. Q. El., Vol. QE-23, March 1985.

Del Bocca, R. L.; Mongeon, R. J.: Multifunction CO_2 Laser Radar Technology. SPIE Proc., Vol. 300, 19-32, Aug. 1981.

Freed, C.: Frequency Stabilization of CO_2 Lasers; Some Applications. SPIE Proc., Vol. 335, 59-68, 1982.

Harper, L. L.; Williams, R. H.; Logan, K. E.; Stevens, D. A. (ILS Inc.) and Degnan, J. J. (NASA/GSFC): Ultrashort Pulse Solid-State Transmitter Development. Proc. Int. Conf. on Lasers 78, Dec. 1978.

Harper, L. L.; Logan, K. E.; Williams, R. H.; Stevens, D. A.: Nd:YAG Development for Spaceborne Laser Ranging System. NASA-CR-159932/NAS5-22916. Final Report Jan. 1976-Feb. 1979.

Hess, R. V.; Brockman, P.; Bair, C. H.; Staton, L. D.; Lytle, C. D. (NASA/LaRC); Laughman, L. M. (CLS Laser Systems Inc.); Kaplan (Systems and Applied Sciences Corp.): Coherent Infrared Lidar Mission and Technology Needs for Measurements of Transport and Concentration of Tropospheric Trace Species. SPIE Proc., Vol. 415, 52-70, April 1983.

Hochuli, U.: Investigation of RF Excited CO_2 Waveguide Lasers. Proposal to J. Degnan, GSFC, June 1983.

Hochuli, U.: Investigation of RF Excited CW CO_2 Waveguide Lasers. Progress Report NASA Grant NAG5-263/ Degnan/GSFC, April 1984.

Kane, T. L.; Zhou, B.; Byer, R. L.: Potential for Coherent Doppler Wind Velocity Lidar Using Neodymium Lasers. Appl. Opt., Vol. 23, No. 11, 2477-2481, 1984.

Kane, T. L.; Byer, R. L.: Doppler Lidar Using Nd:YAG Technology. Optical Remote Sensing of the Atmosphere Technical Digest. Incline Village, Nevada, January 1985.

REFERENCES (continued)

Laughman, L. M.: Long Term Operation of a Sealed Waveguide CO_2 Laser. Rev. Sci. Instrum., Vol. 47, No. 11, 1411-1413, 1976.

Levenson, M. D.; Schawlow, A. L.: Hyperfine Interactions in Molecular Iodine. Physical Review A, Vol. 6, pp. 6-20, 1972.

Rogowski, R. S.; Miller, I. M.; Wood, G. M.; Schryer, D. R.; Hess, R. V. (NASA/LaRC) and Upchurch, B. T. (Chemicon, Inc.): Evaluation of Catalyst for Closed Cycle Operation of High Energy Pulsed Lasers. SPIE Proc., Vol. 415, 112-117, April 1983.

Stark, D. S.; Crocker, A.; Steward, G. J.: A Sealed 100Hz CO_2 TEA Laser Using High CO_2 Concentrations and Ambient Temperature Catalysts (RSRE). J. Phys. E. Sci. Instrum., Vol. 16, 1983.

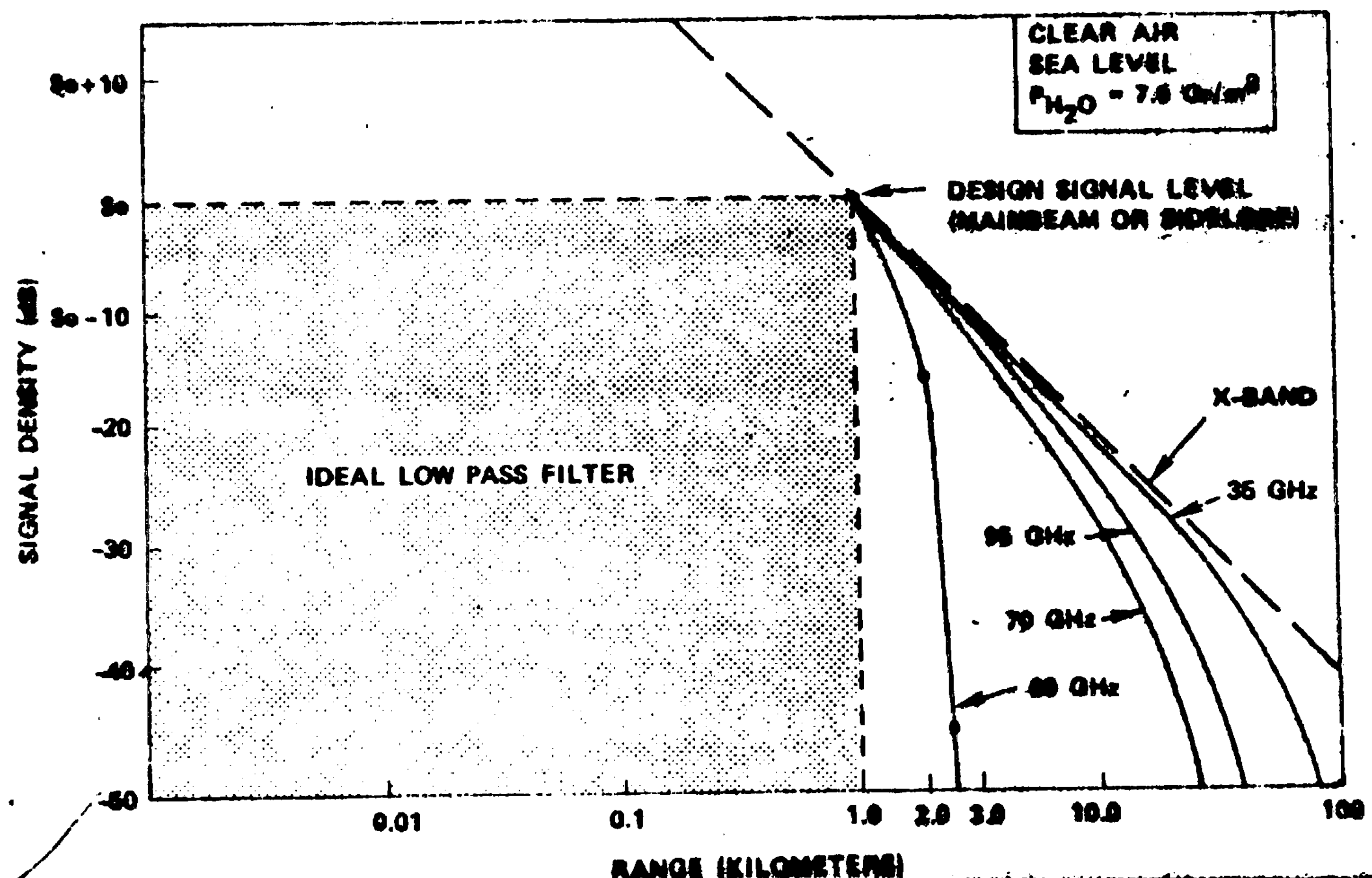
Tanaka, K.; Morinaya, A.: Preliminary Experiment for Stabilizing Unmodulated He-Ne Laser to an Iodine Hyperfine Component. J. Appl. Phys. 50(1), pp. 38-41, 1979.

TRANSCEIVER, FM/CW

HUGHES

SIGNAL "OVERSHOOT" vs FREQUENCY

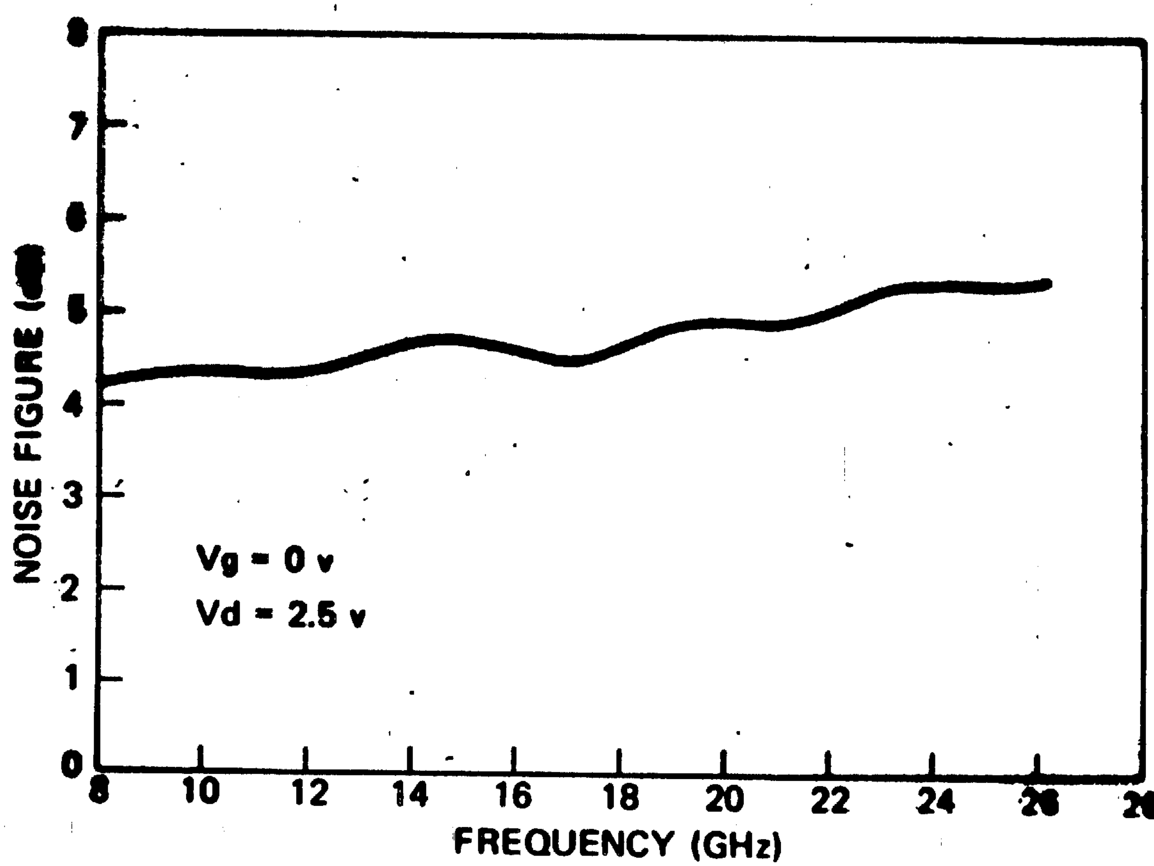
HUGHES



PRESENTATION

REQUESTER/ORIGINATOR DATE VG. NO.

7 — SECTION DISTRIBUTED AMPLIFIER NOISE FIGURE



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HUGHES

SUMMARY OF POWER FET RESULTS

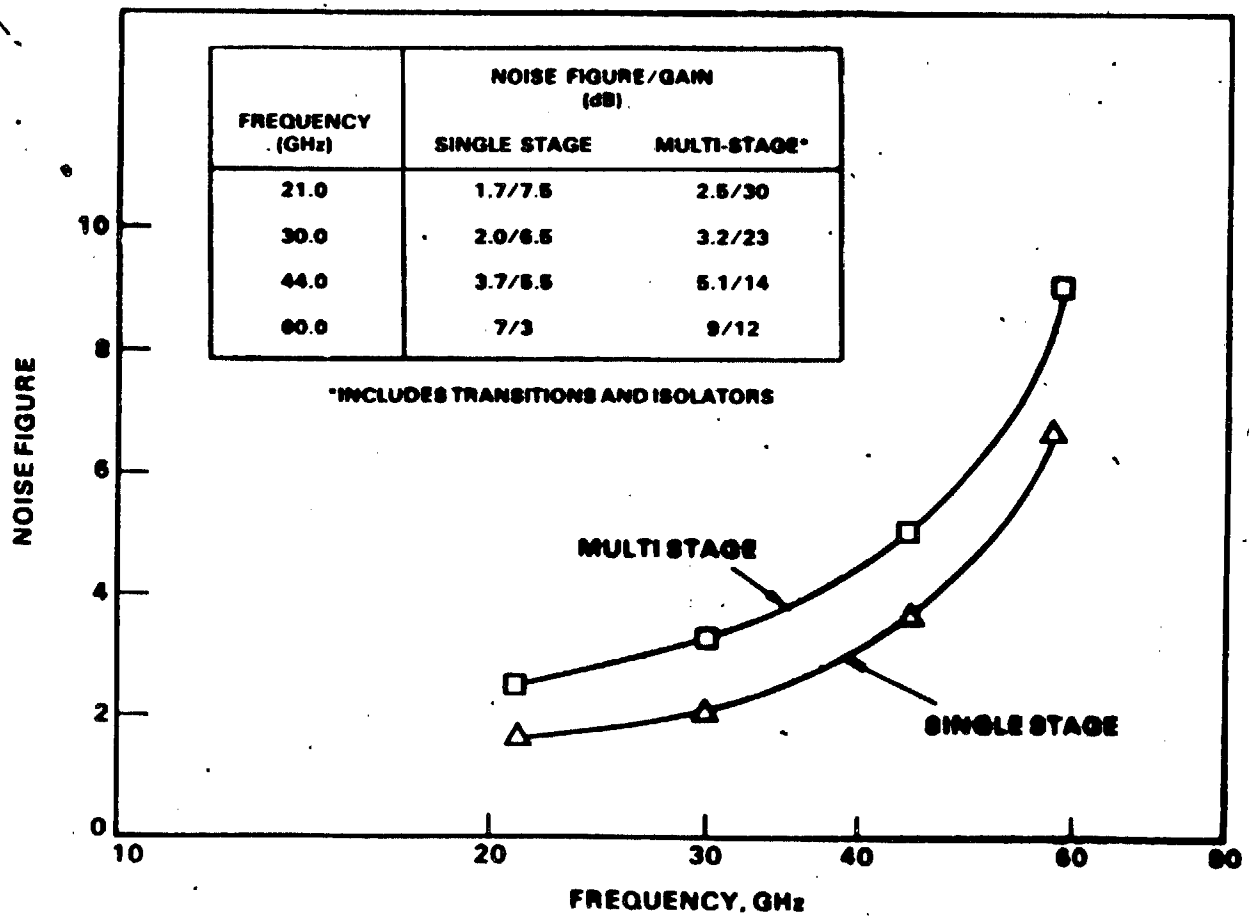
10 GHz	1.8 W BEST 1.5 W TYP	MCC
13 GHz	1.8 W BEST 1.5 W TYP	MCC
15 GHz	2.7 W BEST 2.0 W TYP	CHIP PC
18 GHz	1.25 W BEST	CHIP PC
21 GHz	200 MW BEST 150 MW TYP	MCC
35 GHz	107 MW BEST	AMP
60 GHz	4 MW BEST	AMP

HUGHES FET LNA NOISE PERFORMANCE AT 21, 30, 44 AND 60 GHz

HUGHES

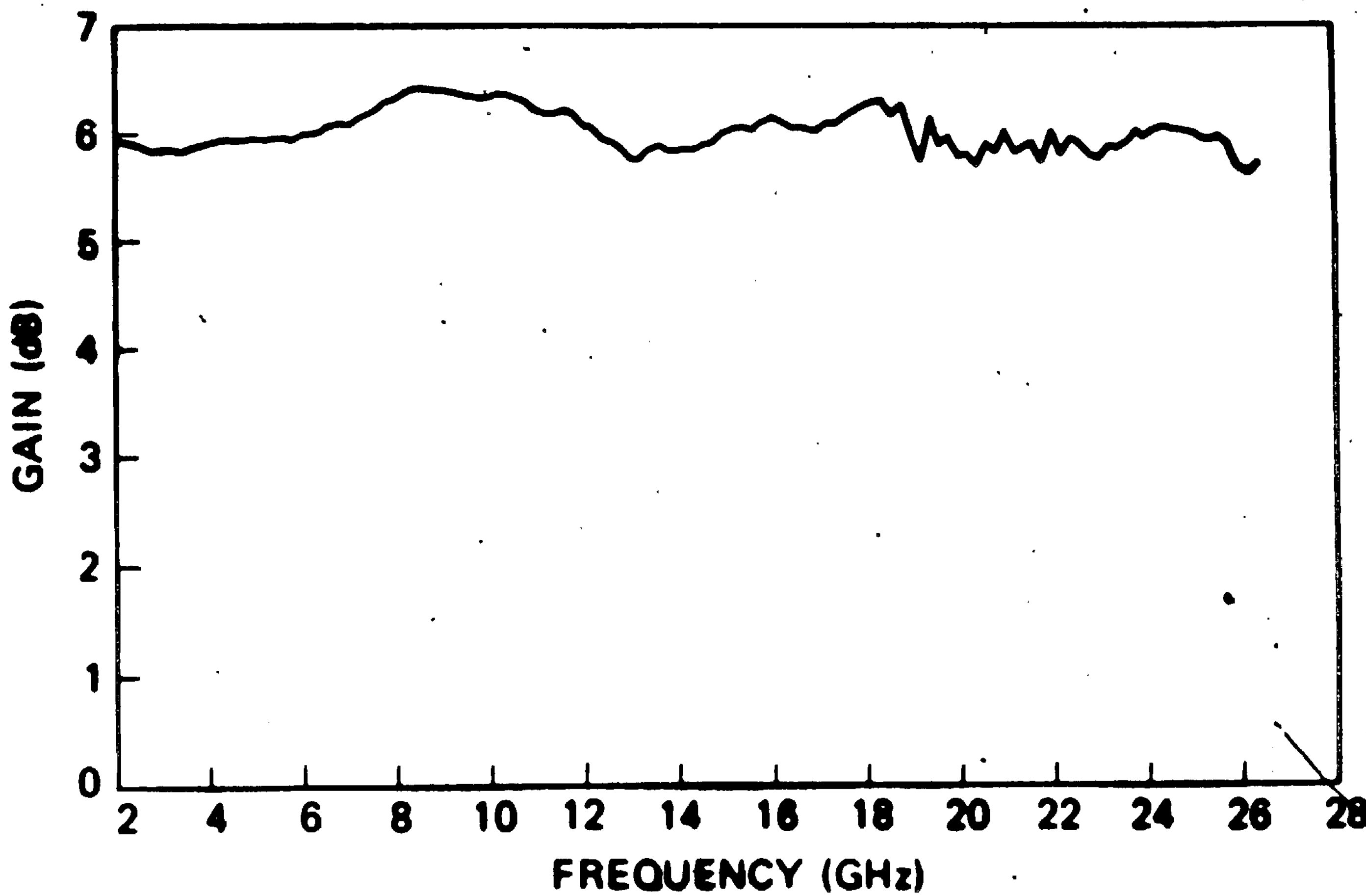
FREQUENCY (GHz)	NOISE FIGURE/GAIN (dB) SINGLE STAGE	MULTI-STAGE*
21.0	1.7/7.5	2.5/30
30.0	2.0/6.5	3.2/23
44.0	3.7/5.5	5.1/14
60.0	7/3	9/12

*INCLUDES TRANSITIONS AND ISOLATORS



7 - SECTION DISTRIBUTED AMPLIFIER

HUGHES



UNCLASSIFIED

HUGHES

PRESENTATION

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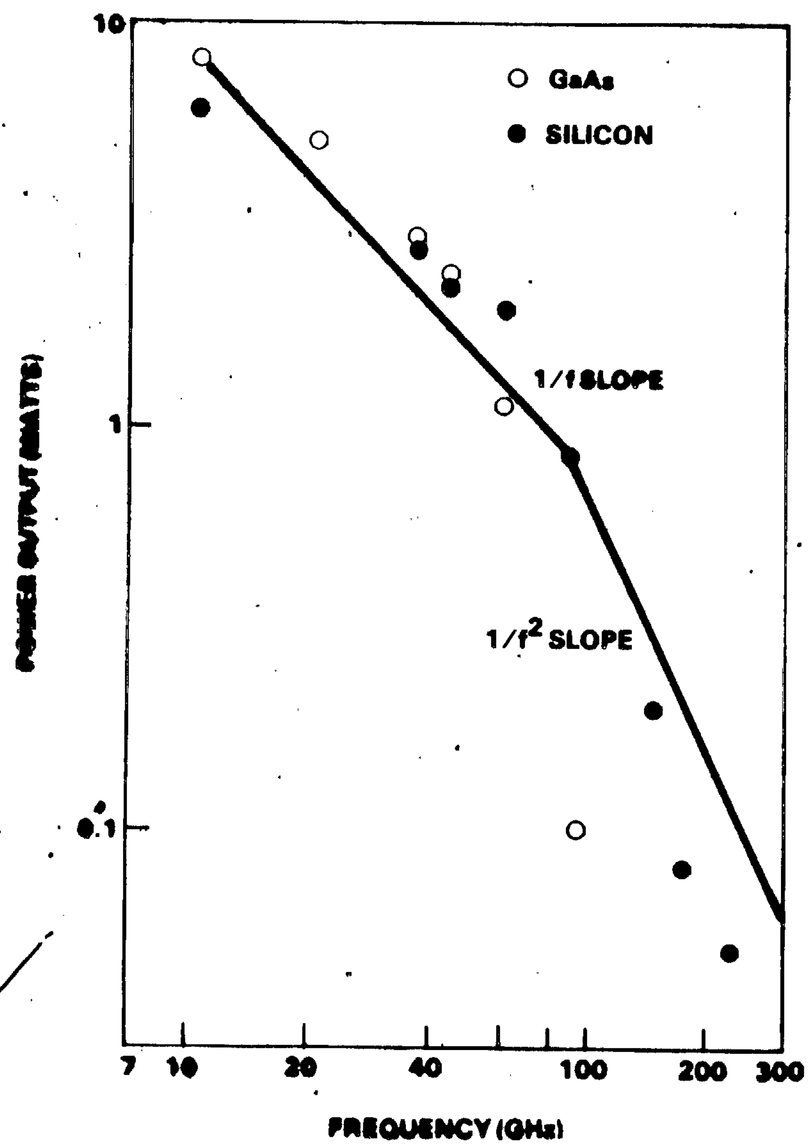
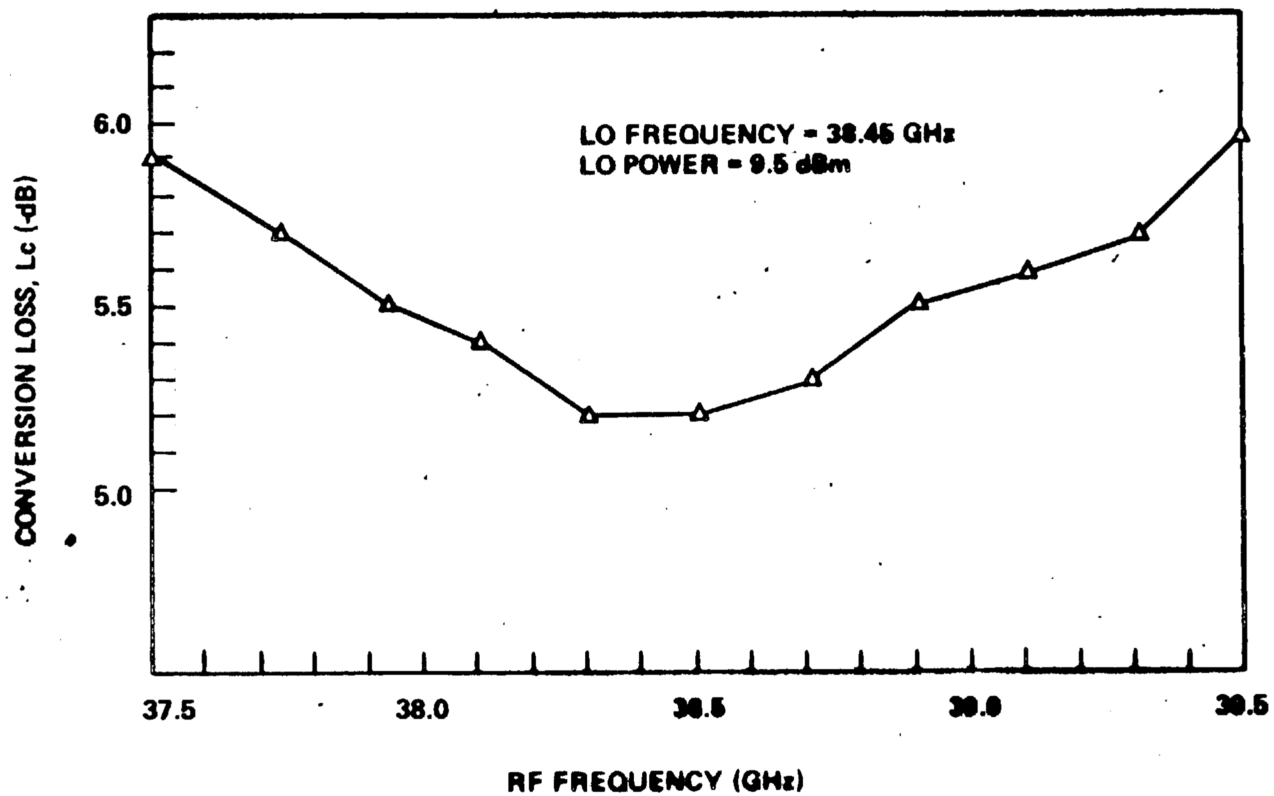


HUGHES

STATE-OF-THE-ART OF CW IMPATT PERFORMANCE

MIXER PERFORMANCE

HUGHES



TRANSCEIVER, FM/CW

HUGHES

UNCLASSIFIED

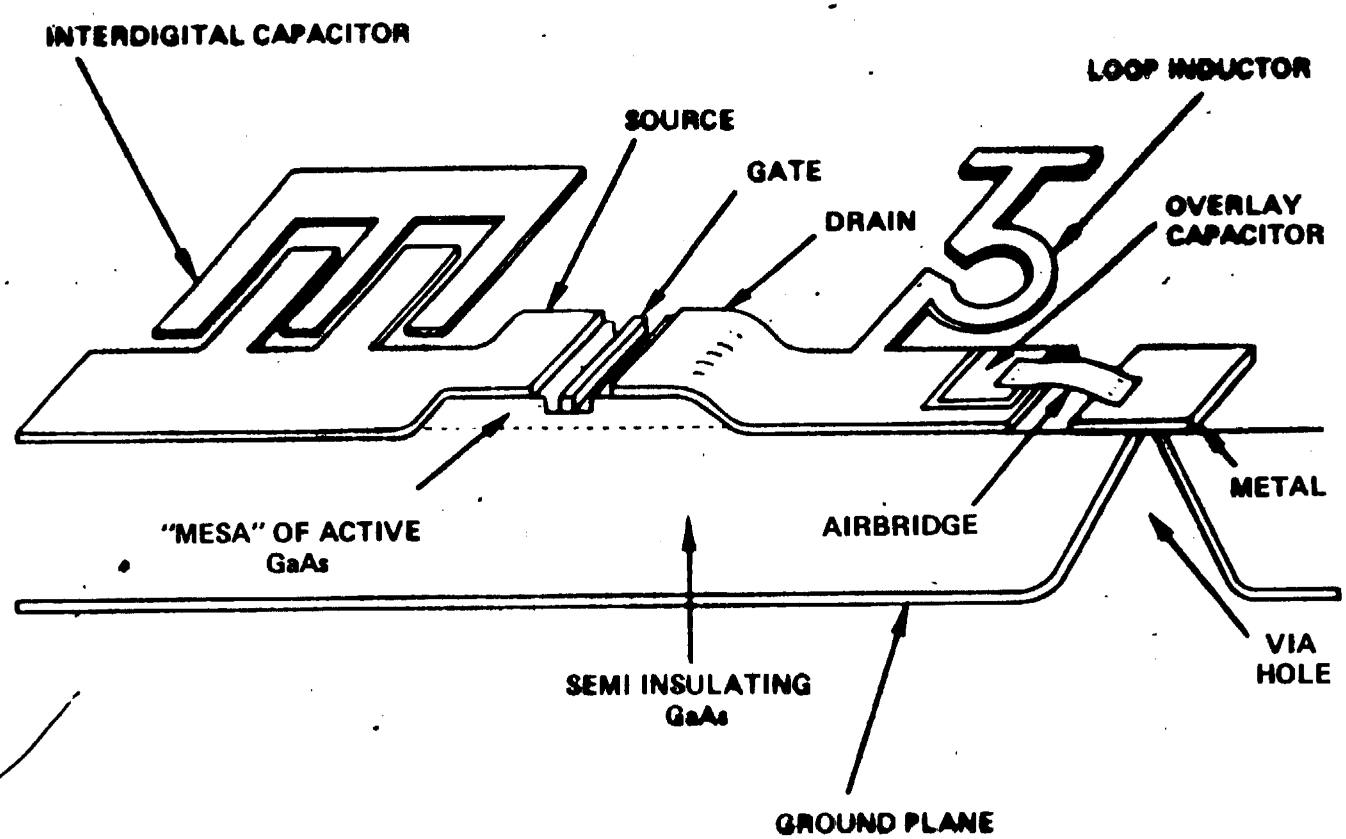
PRESENTATION ______

REQUESTER/ORIGINATOR ______ DATE ______ VG. NO. ______

HUGHES AIRCRAFT COMPANY

MONOLITHIC INTEGRATED CIRCUIT

HUGHES



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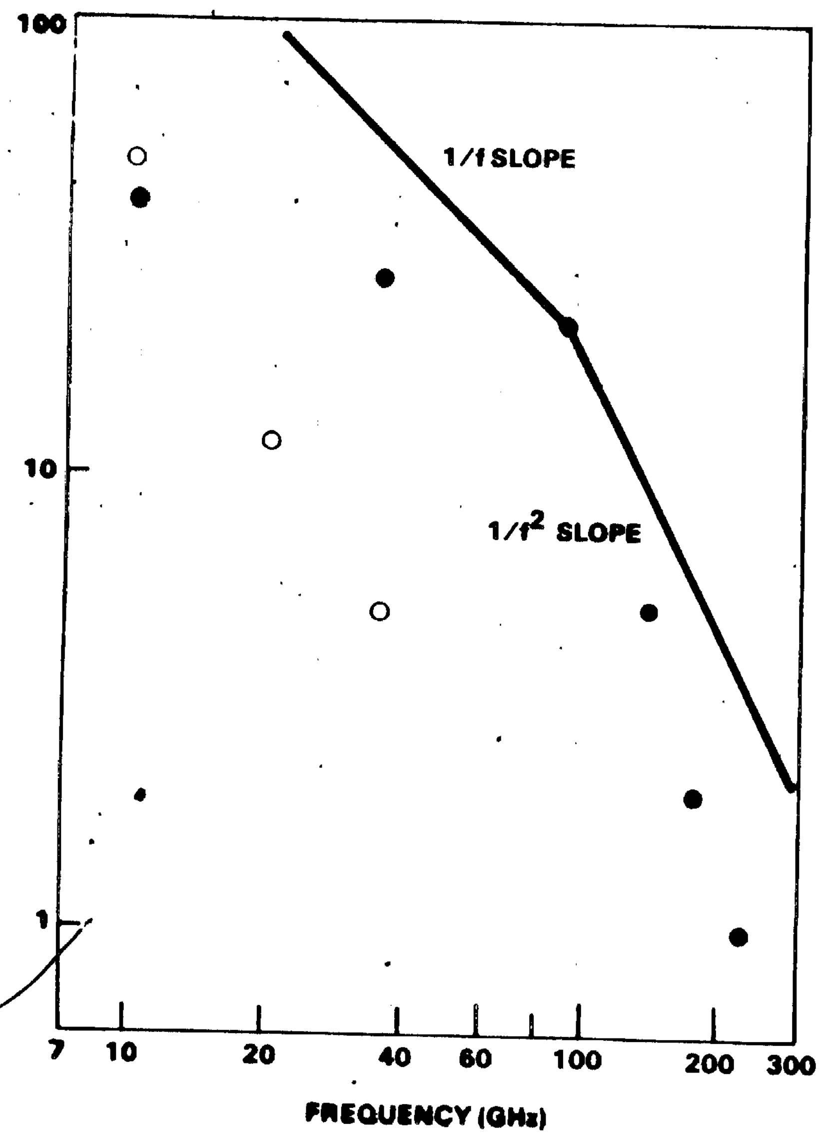
UNCLASSIFIED

PRESENTATION ______

REQUESTER/ORIGINATOR ______ DATE ______ VG. NO. ______

HUGHES



HUGHES

STATE-OF-THE-ART OF PULSED IMPATT PERFORMANCE

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ORIGINALS FOR

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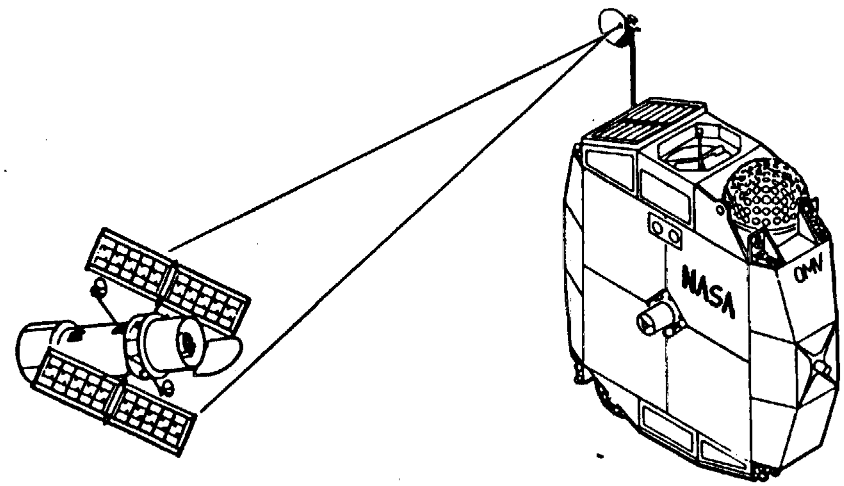
A MILLIMETER WAVE RANGE/RANGE RATE SYSTEM FOR OMV APPLICATION



PRESENTED BY

E. R. FEAGLER
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ALLIED BENDIX AEROSPACE
GUIDANCE SYSTEMS DIVISION
400 SOUTH BEIGER STREET
MISHAWAKA, INDIANA 46544

FEBRUARY 20, 1985

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ABSTRACT

The Orbital Maneuvering Vehicle (OMV) is a general utility spacecraft designed to provide delivery, retrieval, and servicing of other spacecraft. A majority of the OMV missions envisioned require rendezvous and docking maneuvers which are coordinated through a ground station, utilizing a satellite tracking network and OMV on-board sensors. Of particular concern herein, is an on-board sensor which aids the OMV in performing rendezvous and docking maneuvers by providing measurements of range, range rate, and bearing to the spacecraft target. A millimeter wave sensor is described which utilizes a spread spectrum random noise waveform, solid-state transmitter, and correlation processing, to provide target acquisition and tracking, from initial acquisition at a maximum range of 12 nmi, to docking at essentially zero range.

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OMV RANGE/RANGE RATE SENSOR

PERFORMANCE GOALS

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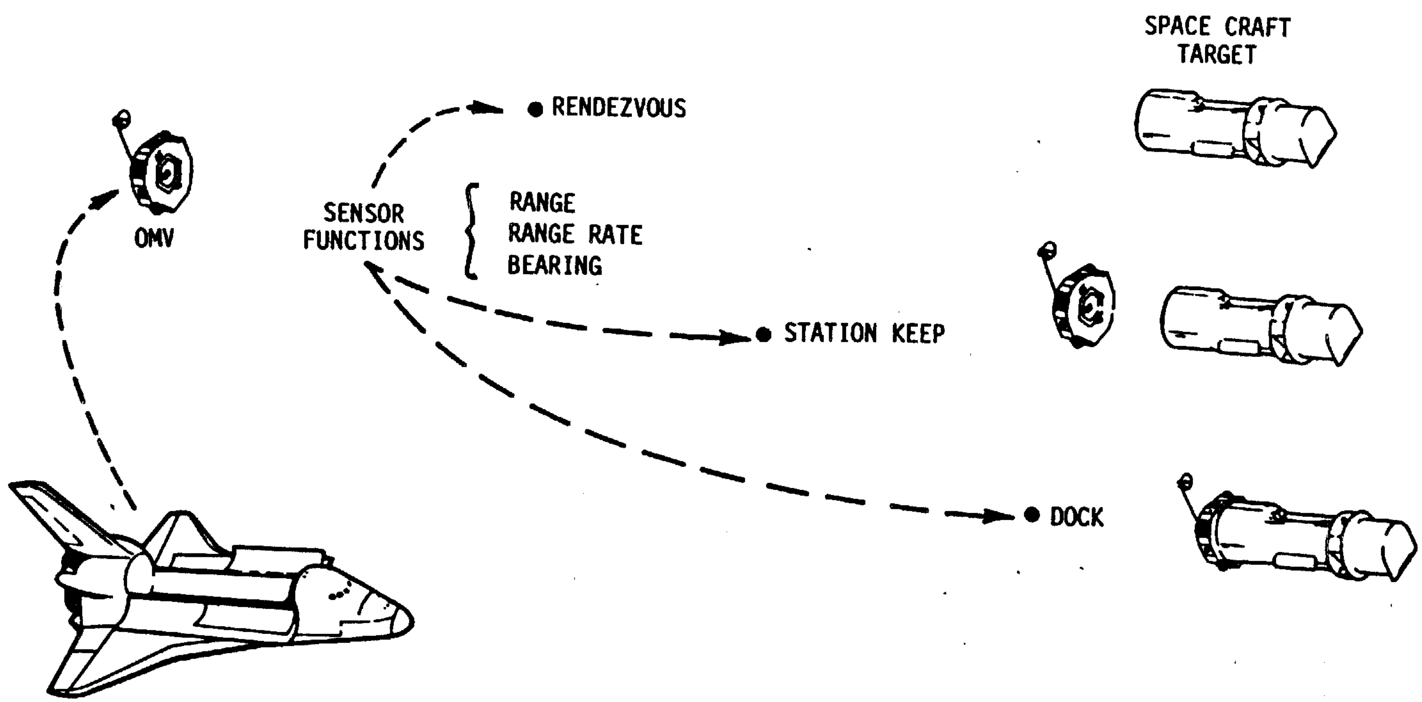
OMV APPLICATIONS OVERVIEW

The Orbital Maneuvering Vehicle (OMV) operating from the Space Shuttle will be capable of payload delivery, retrieval, and maintenance to altitudes as high as 2000 nmi. The OMV is also expected to provide round trip logistics supply service to the Space Station to support assets in other orbits. During a typical mission the OMV is required to rendezvous, stationkeep, and dock with the spacecraft target.

An on-board sensor is being proposed which would aid in the initial rendezvous maneuvers by providing range to the spacecraft target and closing rate as well as the target bearing. During stationkeeping and final docking maneuvers the sensor will continue to provide relative position and rate information to essentially zero range. The range and range rate information is likewise available for updating and as a backup for the inertial navigation system.

OMV APPLICATIONS OVERVIEW



TYPICAL
<u>OMV MISSIONS</u>

- DELIVERY
- RETRIEVAL
- SERVICE
- REBOOST

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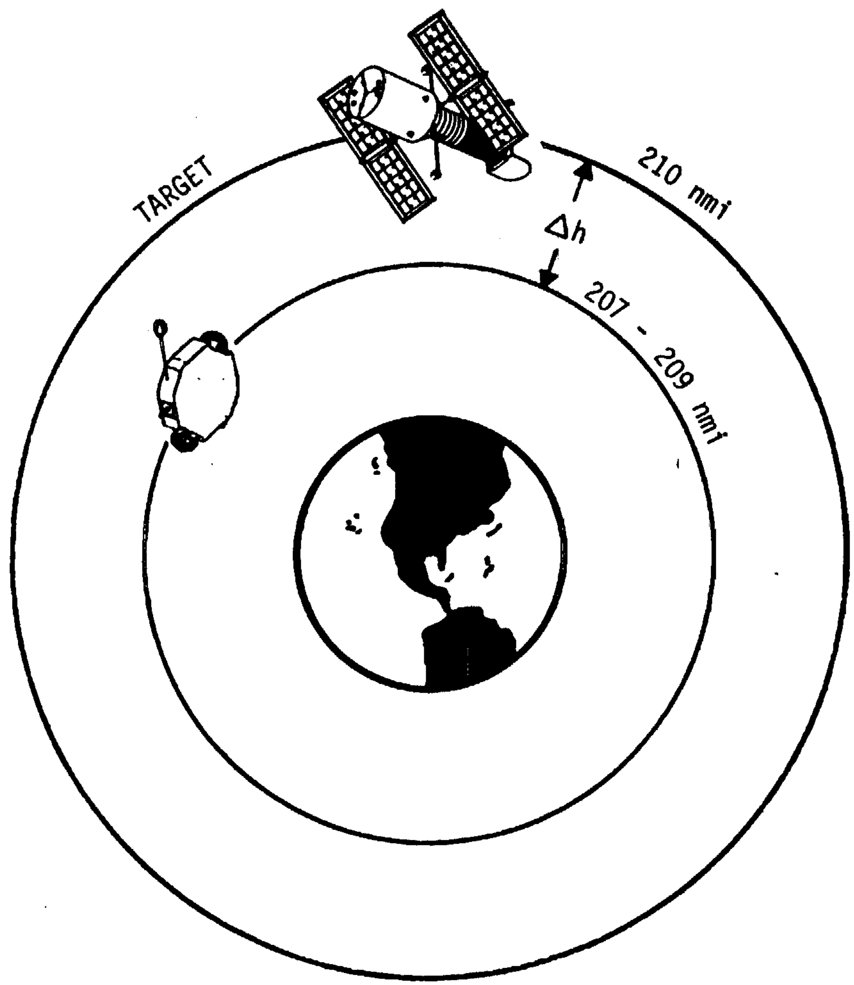
OMV - TARGET RELATIVE GEOMETRY

The relative geometry between the OMV and spacecraft target during initial rendezvous maneuvers establishes the search requirements for the OMV on-board Range/Range Rate (R/R) Sensor. As an example of a typical OMV mission, consider the requirement to service a spacecraft target which is in a 210 nmi circular orbit.

At initiation of the target rendezvous phase it will be assumed that the OMV has maneuvered into a coplanar circular orbit at an altitude differential on the order of 1 to 3 nmi below the target. When the median range to the target is 8 nmi the Range/Range Rate (R/$\dot{R}$) Sensor begins an angle, range, and doppler search for the target which is located within an error ellipse with dimensions ± 4 nmi T (tangential), ± 1 nmi R (radial) and ± 1/2 nmi N (normal) as specified by NASA MSFC. It is assumed that an on-board GPS receiver is used to accurately establish the OMV position. The sensor parameters to be defined for this relative geometry are the angle and range coverage required as functions of search time.

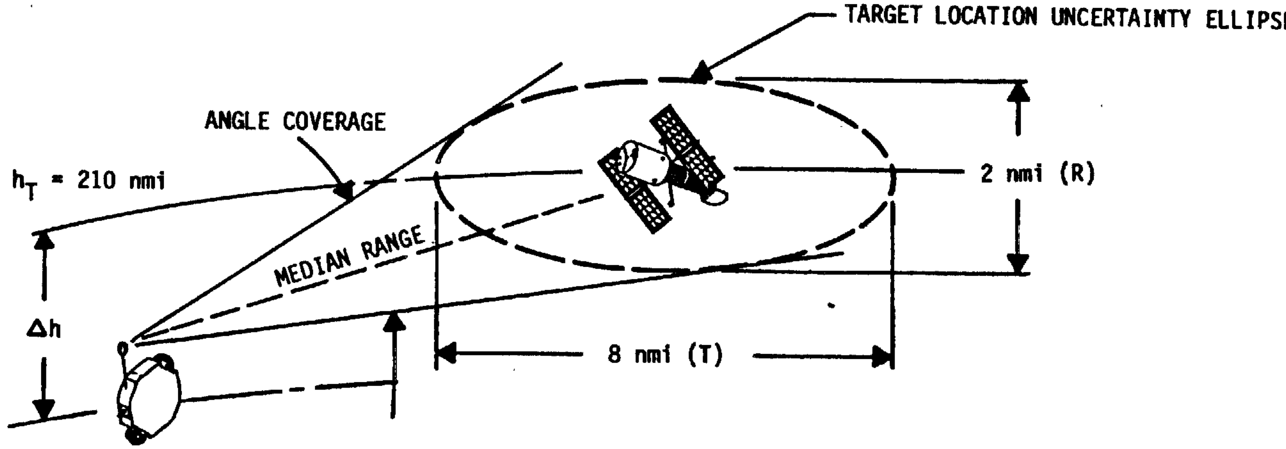
OMV - TARGET RELATIVE GEOMETRY



OMV R/$\dot{R}$ SENSOR INITIAL SEARCH PARAMETERS

During the rendezvous maneuver the R/$\dot{R}$ sensor is required to search over the target location error ellipse which extends 8 nmi in range. The corresponding angle coverage required is a function of the OMV-spacecraft target altitude differential, Δh. The required angle coverage and median range were established as functions of time for a target which was located in a circular orbit at an altitude of 210 nmi. It was assumed that the median range at initiation of target search was 8 nmi and the maximum angle coverage provided by the R/$\dot{R}$ sensor was 30° included angle.

OMV R/$\dot{R}$ SENSOR INITIAL SEARCH PARAMETERS



ASSUMPTIONS

- TARGET IN 210 nmi CIRCULAR ORBIT
- OMV IN A CIRCULAR ORBIT Δh (1-3) nmi BELOW TARGET
- TARGET LOCATION UNCERTAINTY ELLIPSE:
  ±4 nmi (T); ±1 nmi (R); ±0.5 nmi (N)
- MEDIAN RANGE AT INITIATION OF SEARCH IS 8 nmi
- R/$\dot{R}$ SENSOR ANGLE AND RANGE COVERAGE 30° BY 8 nmi

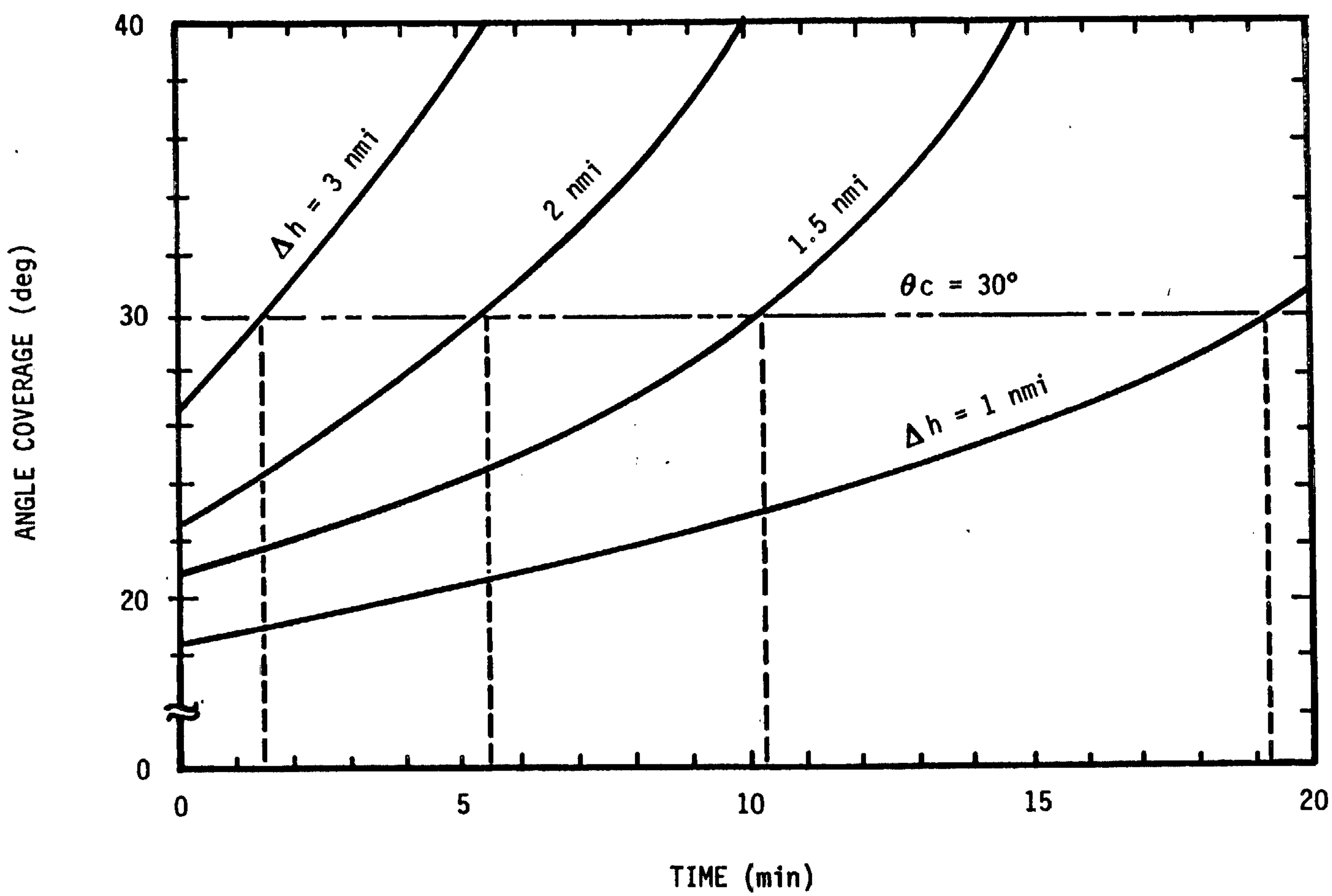
TO BE ESTABLISHED

- MEDIAN RANGE AND ANGLE COVERAGE VS TIME

REQUIRED ANGLE COVERAGE VS TIME

The search time corresponding to 30° angle coverage is inversely proportional to the altitude differential, Δh, between the target and OMV. As the altitude differential decreases from 3 nmi to 1 nmi, the time available to perform the target search increases from 1.5 minutes to approximately 20 minutes. An altitude differential of 1.5 nmi the search time corresponds to a midrange value of approximately 10 minutes. As will be shown, search times on the order of 10 minutes enable configuring a sensor with a relatively low power, solid-state transmitter.

REQUIRED ANGLE COVERAGE VS TIME



MEDIAN RANGE VS TIME

At initiation of target search the median range to the target location uncertainty ellipse is 8 nmi and the closure rate, expressed in nautical miles per minute, is equal to one-tenth the OMV-target altitude differential. Consequently, during the 10 minutes available to search an 8 nmi by 30° sector at an altitude differential of 1.5 nmi, the median target range is reduced by 1.5 nmi to 6.5 nmi. Initiating the target search at closest edge of the target error ellipse would assure early detection of closest targets and allow maximum time to maneuver into position for docking.

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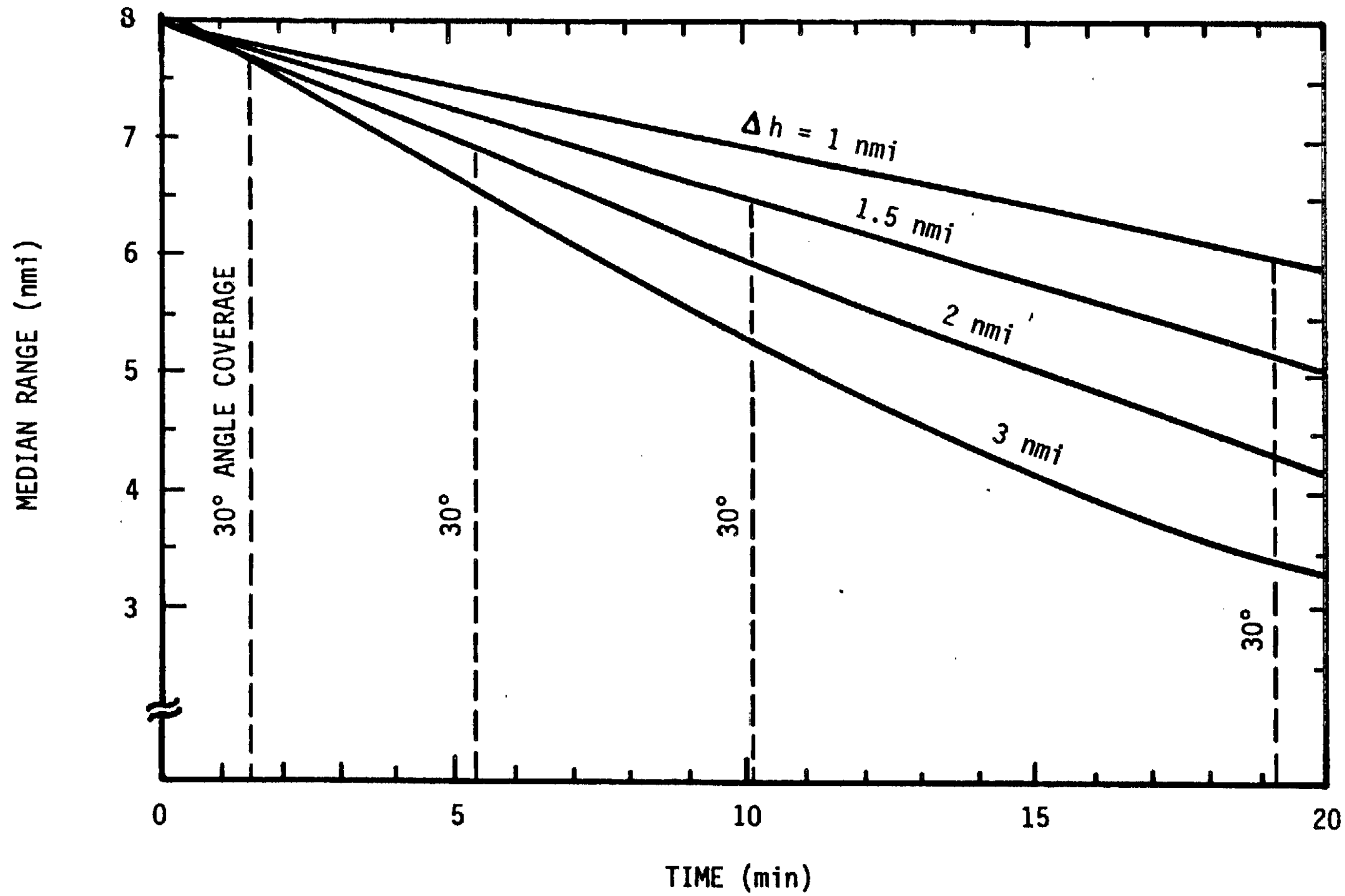
MEDIAN RANGE VS TIME



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OMV RANGE/RANGE RATE SENSOR
PERFORMANCE GOALS (PRELIMINARY)

A preliminary set of Range/Range Rate (R/$\dot{R}$) sensor performance goals have been established in conjunction with NASA MSFC and OMV prime contractors. Two levels of performance have been specified, a high accuracy mode for close in ranges of 100 ft. or less, and a lower accuracy mode for ranges greater than 100 ft. At the longer ranges where initial acquisition occurs, the primary concern is providing the search coverage required. At the closer ranges while docking is proceeding the accuracy is of prime concern. The preliminary performance goals were used as a guide in formulating the baseline R/$\dot{R}$ sensor design approach.

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OMV RANGE/RANGE RATE SENSOR
PERFORMANCE GOALS (PRELIMINARY)

- RANGE COVERAGE - 0 TO 12 NAUTICAL MILES (NMI)
- RANGE ACCURACY
  - 0 TO 100 FT. RANGE: ± 6 INCHES OR 1% OF RANGE
  - 100 FT. TO 12 NMI: ± (20 FT. + 2% OF RANGE)
- RANGE RATE COVERAGE - 0 TO ± 100 FT./SEC.
- RANGE RATE ACCURACY
  - 0 TO 100 FT. RANGE: ± 0.1 FT./SEC.
  - 100 FT. TO 12 NMI: ± 5% OF RANGE RATE
- ANGLE COVERAGE - 30° INCLUDED CONICAL AREA
- SEARCH PARAMETERS - 30° BY 8 NMI SECTOR, MAX. SEARCH TIME 10 MIN.
- TARGET DETECTION - 99% DETECTION PROBABILITY FOR $50 M^2$ TARGET AT 12 NMI
- WEIGHT/VOLUME/POWER - DESIRABLE TO BE 50 LBS. OR LESS, 1 $FT.^3$ OR LESS AND MINIMUM USE OF 28 VDC POWER

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RANGE/RANGE RATE SENSOR DESIGN APPROACH

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BASELINE OMV R/$\dot{R}$ SENSOR DESCRIPTION

In selection of the baseline approach major emphasis was placed on achieving the required performance goals with a sensor configuration which minimized size, weight, and primary power requirements. By utilizing the high processing gains achievable with the Bendix Random Signal Radar (RSR) spread spectrum techniques, a high duty cycle-low power solid-state implementation is achievable with performance which satisfies design goals.

The R/$\dot{R}$ sensor operates in a free space environment, therefore advantage is taken of smaller aperture requirements afforded by operating at the higher millimeter wave frequencies. A second advantage of operating at millimeter wave frequencies is the increased isolation achievable between antenna pairs for CW mode operation.

The RSR transmitter modulation function is derived by sampling truly random noise which is not time repetitive. As a consequence range measurements are unambiguous whether the radar is implemented for CW operation with two antennas or single antenna Interrupted CW (ICW) operation. By combining ICW and CW modes of operation continuous range coverage is provided from initial acquisition at a maximum range of 12 nmi to docking at essentially zero range.

An advantage of the digital signal processing utilized by the RSR is the ease with which range and Doppler resolutions are modified to adapt to changing performance requirements.

In addition to providing reliable operation, the solid-state millimeter wave R/$\dot{R}$ Sensor design approach minimizes size, weight, and power requirements.

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BASELINE OMV R/Ṙ SENSOR DESCRIPTION

APPROACH:

- MILLIMETER WAVE ADAPTATION OF BENDIX RANDOM SIGNAL RADAR COMBINING INTERRUPTED CW AND CW MODES OF OPERATION.

ADVANTAGES:

- DUAL OPERATIONAL MODE PROVIDES CONTINUOUS RANGE COVERAGE FROM INITIATION OF RENDEZVOUS TO DOCKING.
- RANDOM CODE PROVIDES UNAMBIGUOUS RANGE MEASUREMENTS.
- HIGH RANGE AND DOPPLER RESOLUTIONS ARE READILY IMPLEMENTED TO ACHIEVE SPECIFIED ACCURACIES.
- RELIABLE OPERATION PROVIDED BY SOLID-STATE COMPONENTS.
- SOLID-STATE MILLIMETER WAVE APPROACH MINIMIZES SIZE, WEIGHT, AND POWER REQUIREMENTS.

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RANDOM SIGNAL RADAR CONCEPT

The Bendix Random Signal Radar (RSR) is a spread spectrum radar which transmits a low power, bandwidth limited, random noise waveform. The signal received from the target is cross-correlated with a sample of the transmitted signal, effectively compressing the spread spectrum waveform into a narrow-band signal. The resulting narrow-band signal is then processed to provide independent measurements of target range, range rate, and bearing.

The transmitted spread spectrum waveform, $s(t)$, is generated by biphase modulating a carrier with a binary code, $m(t)$, derived from sampled random noise. The target return signal, $a_K\ s(t-T_K)$, is an attenuated and delayed replica of the transmitted signal with a delay proportional to target range.

The carrier is removed from the return signal and the resulting video signal plus receiver noise, $a_K\ s(t - T_K) + n(t)$, is cross-correlated with an appropriately delayed sample of the transmitter modulation code, $m(t-T_K)$. As a result of the cross-correlation process the target return signal which at this point is a broad-bandwidth signal is compressed into a narrow-band signal. By filtering the negative signal-to-noise ratio at the video limiting amplifier output is increased to the positive level required for signal processing. The limiting amplifier provides instantaneous gain control and responds as a linear amplifier for the signal which is below receiver noise power. Digital techniques perform the cross-correlation processing simply and cheaply.

One of the products of the multiplication in the excluse OR is a sinusoid at a frequency corresponding to the rate of change of range (Doppler) between the sensor and target. This signal is extracted from receiver noise by the band-pass filter at the output of the exclusive-OR multiplier. Angle, range, and Doppler functions present at the correlator output are processed by a microcomputer and are available as data inputs to the OMV.

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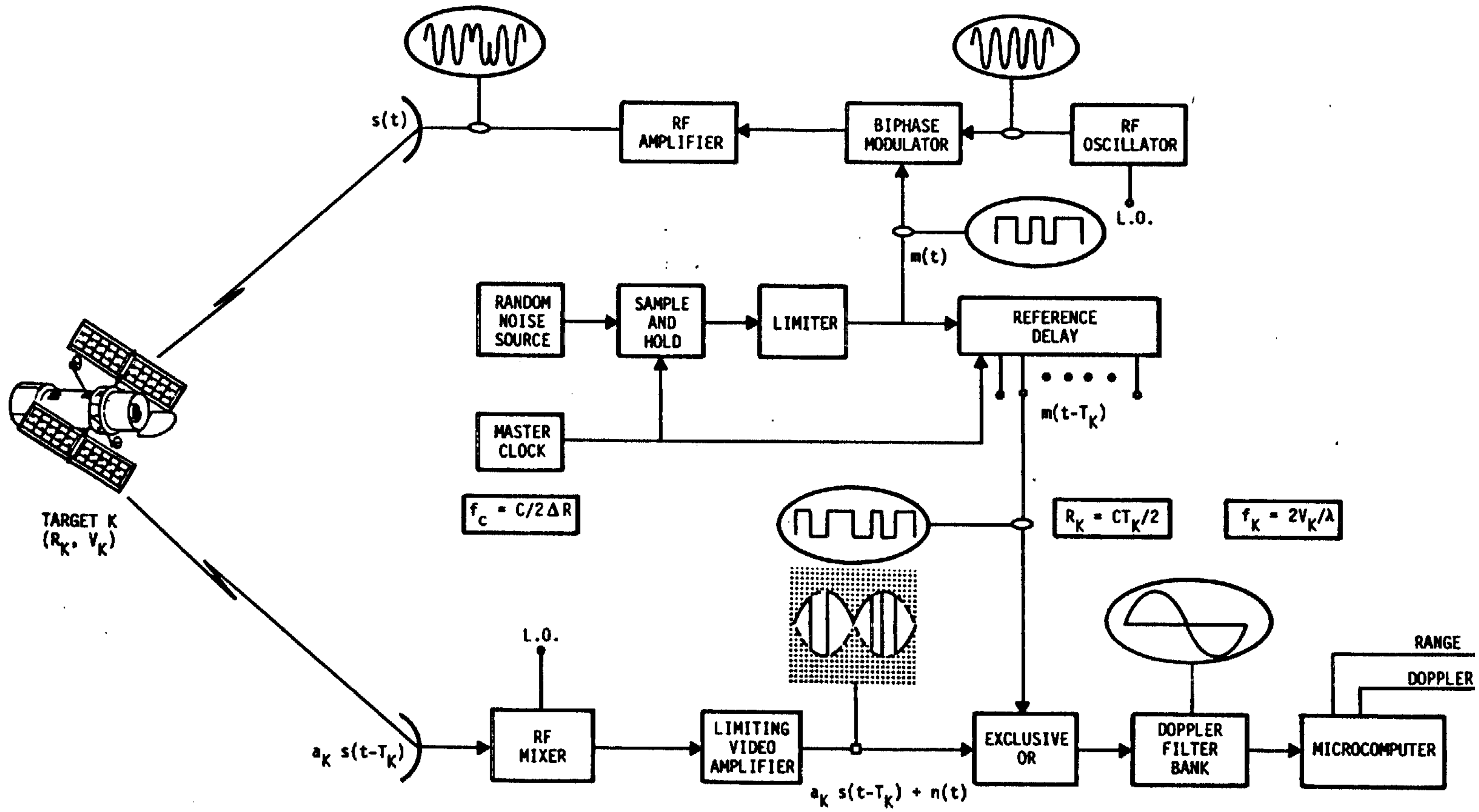
RANDOM SIGNAL RADAR CONCEPT



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RANDOM SIGNAL RADAR RESOLUTION AND CORRELATION GAIN

The RSR can be configured to provide any arbitrary combination of range and Doppler resolutions. The individual resolutions are selected to provide required system performances in terms of measurement accuracies and tracking loop dynamics, and the combination defines the radar processing (correlator) gain.

Range resolution, which is inversely proportional to the transmitted bandwidth, is normally selected to provide a required range measurement accuracy. The Doppler resolution in addition to establishing the separation between Doppler frequencies distinguishable, likewise defines the coherent integration time of the correlation process (the integration time is equal to the reciprocal of the Doppler resolution).

The ratio of the transmitted bandwidth to the Doppler bandwidth is equal to the time-bandwidth product of the transmitted waveform. The correlator gain which is proportional to the time-bandwidth product defines the ratio of output-to-input signal-to-noise ratio.

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RANDOM SIGNAL RADAR RESOLUTION AND CORRELATOR GAIN

RANGE RESOLUTION (ΔR):

- INVERSELY PROPORTIONAL TO TRANSMITTED NOISE BANDWIDTH,

$$\Delta R = C/2B_N$$

DOPPLER RESOLUTION (Δf_d):

- INVERSELY PROPORTIONAL TO COHERENT INTEGRATION TIME,

$$\Delta f_d = 1/T_O$$

CORRELATOR GAIN (G_C):

- PROPORTIONAL TO TIME-BANDWIDTH PRODUCT,

$$G_C \sim T_O B_N$$

- DEFINES RATIO OF $(S/N)_O/(S/N)_I$,

$$(S/N)_O = G_C (S/N)_I$$

TRANSMITTED SIGNAL (BIPHASE MODULATION)

The RSR biphase transmitted waveform has a power spectral density function with a $(\sin X/X)^2$ envelope characteristic. Envelope nulls occur at a frequency corresponding to the sampling clock frequency and the time differential, Δt, is equal to the reciprocal of the clock frequency.

The autocorrelation function for the biphase modulation process has a trangular envelope with the peak occurring at the delay, τ_K, corresponding to the target round trip delay. The equivalent time delay width of the correlation function is inversely proportional to the transmitted signal bandwidth and defines the system range resolution. The correlation envelope is processed to provide range acquisition and tracking functions.

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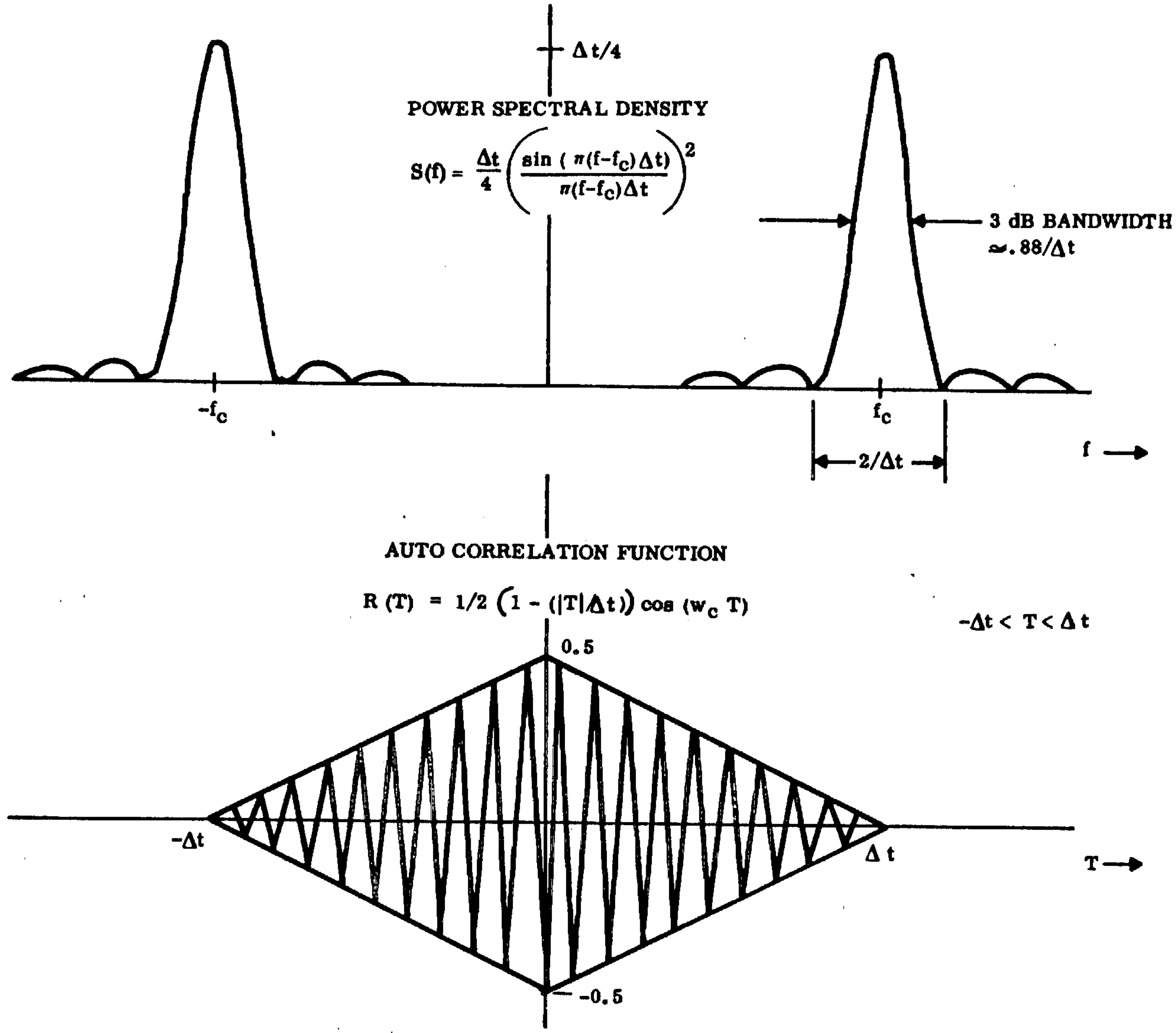
TRANSMITTED SIGNAL (BIPHASE MODULATION)



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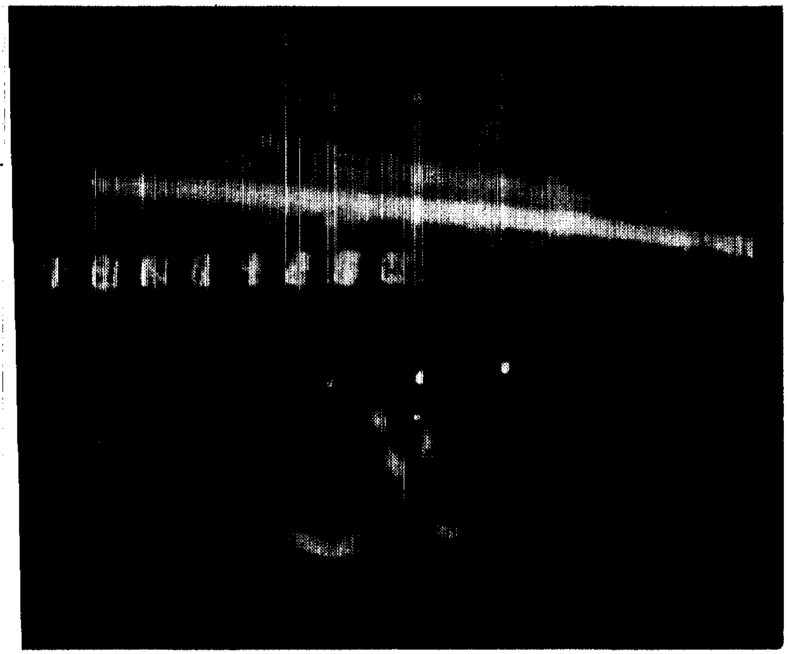
RESOLUTION OF THREE CORNER REFLECTORS

The resolution of three corner reflectors, a $50M^2$ and two $30M^2$ reflectors, is illustrated for separations of three feet and four feet respectively. The measurements were made with a MMW Random Signal Radar brassboard operating at 94 GHz and implemented with a range resolution of 0.8M (2.6 ft.).

With a separation of three feet between reflectors, the returns from the $50M^2$ and the first $30M^2$ corner reflectors are merged illustrating the coherent processing properties of the RSR waveform. Since the receiver processing is linear the relative amplitude and phase relationships between individual scattering points are preserved.

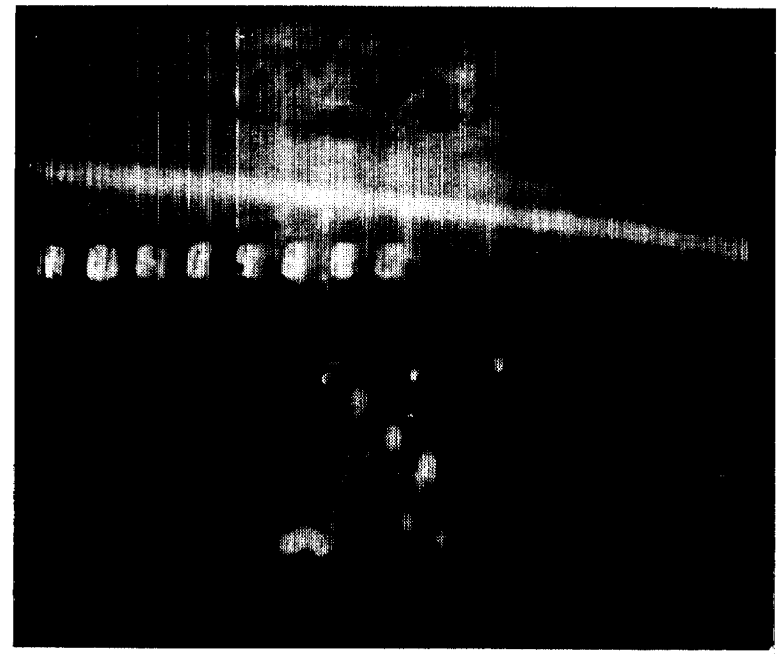
When the corner reflectors are separated by four feet, the individual triangular response characteristics are evident. At a separation of four feet, the response of a particular reflector in adjacent range bins is negligible.

RESOLUTION OF THREE CORNER REFLECTORS



RANGE

THREE FOOT RANGE INTERVALS



RANGE

FOUR FOOT RANGE INTERVALS

RSR BREADBOARD OPERATING CHARACTERISTICS

The operating characteristic of an RSR breadboard implemented with a correlator gain of 53 dB was measured in terms of the output-to-input signal-to-noise ratios. For the normal mode of operation the input signal-to-noise ratio (SNR) is less than unity, and the relationship between output and input SNR's is a linear function. In this linear region of operation the ratio between output and input SNR is equal to the correlator gain. For the breadboard implementation an input SNR of -33 dB results in 20 dB SNR at the correlator output.

The correlator gain likewise establishes the instantaneous dynamic range of the correlator. As the input SNR approaches unity the relationship between output and input SNR becomes nonlinear and a saturation level equal to the correlator gain is approached as the signal power exceeds the receiver noise power. The limiting amplifier and digital signal processing provide an instantaneous gain control which normalizes the correlator output power. When the input signal power is below receiver noise power the signal is AGC'd to the receiver noise level and the signal transfer characteristic is linear. The instantaneous dynamic range is increased by corresponding increases in correlator gain.

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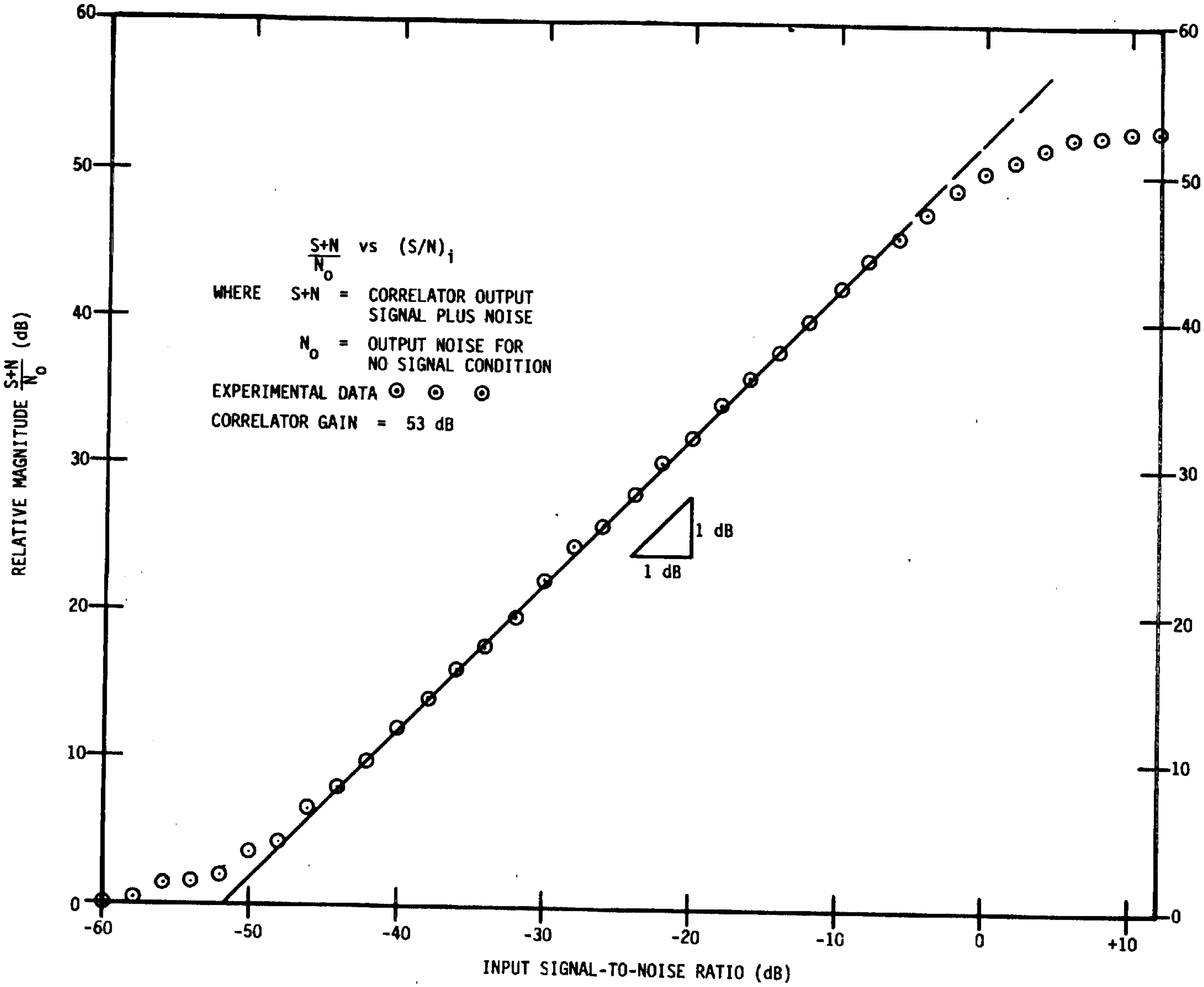
RSR BREADBOARD OPERATING CHARACTERISTICS



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RSR SIGNAL-TO-INTERFERENCE RATIO

A general expression defining the operating characteristics of the RSR includes the effects of sources of interference such as clutter and jamming power when present. As can be seen from the general expression, when the interference power level approaches the receiver noise power at the correlator input, the resulting signal-to-interference ratio is degraded.

Clutter represents RSR transmitted power which is returned from objects other than the target which are either in the range-doppler cell of the target (correlated clutter) or outside the target range-doppler cell (uncorrelated clutter). Correlated clutter competes directly with the target (signal) return, whereas uncorrelated clutter which occurs outside the target range-doppler cell is effectively reduced by the correlator gain.

Jamming power which originates from sources other than the RSR transmitter is processed as a source of interfering noise and is effectively reduced by the correlator gain just as uncorrelated clutter and receiver noise power are.

The particular combination of interfering powers and resulting signal-to-interference ratio are investigated for each application to define what is significant in determining the resulting signal-to-interference ratio.

RSR SIGNAL-TO-INTERFERENCE RATIO

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GENERAL EXPRESSION

$$(S/I)_O = \frac{G_C\, S_I}{S_I + N_I + Cu_I + G_C Cc_I + J_I}$$

$$= \frac{G_C\, (S/N)_I}{I + (S/N)_I + (Cu/N)_I + G_C\, (Cc/N)_I + (J/N)_I}$$

WHERE
- G_C = CORRELATOR GAIN
- $(S/N)_I$ = SIGNAL-TO-NOISE RATIO AT CORRELATOR INPUT
- $(Cu/N)_I$ = UNCORRELATOR CLUTTER-TO-NOISE RATIO
- $(Cc/N)_I$ = CORRELATED CLUTTER-TO-NOISE RATIO
- $(J/N)_I$ = JAMMER-TO-NOISE RATIO

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RANGE/RANGE RATE SENSOR
PARAMETER SELECTION RATIONALE

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FREQUENCY CONSIDERATIONS

For the OMV space application propagation losses are assumed to be zero, consequently the aperture required to achieve a given sensitivity is inversely proportional to the operating frequency. A plot of the aperture required to detect a $50M^2$ target at the maximum R/R sensor operating range of 12 nmi reveals a minimum of 40 cm (16 in) at an operating frequency of 94 GHz. The transmitter power assumed for the designated frequencies is that available from a solid-state source as reported in the Reference listed below.

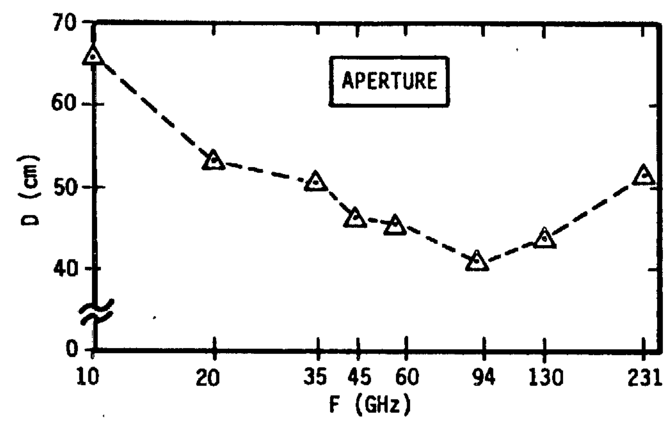
The transmitter primary power (d-c input power) required to generate the transmitted output power is a function of the operating frequency and efficiency of the solid-state device available at that frequency. Both the available output power and device efficiency decrease with increasing frequency. These factors result in a region between 35 GHz and 135 GHz of relatively constant primary power requirements which are on the order of 15 watts.

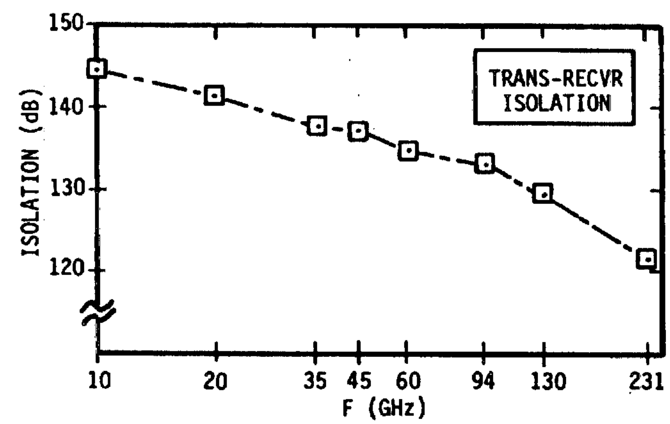
The isolation between the transmitter and receiver signal paths required to prevent degradation in receiver sensitivity decreases as the transmitter power decreases with frequency. At an operating frequency of 94 GHz an isolation of 133 dB is required to maintain transmitter leakage power at a level 10 dB below receiver noise power and prevent performance degradation.

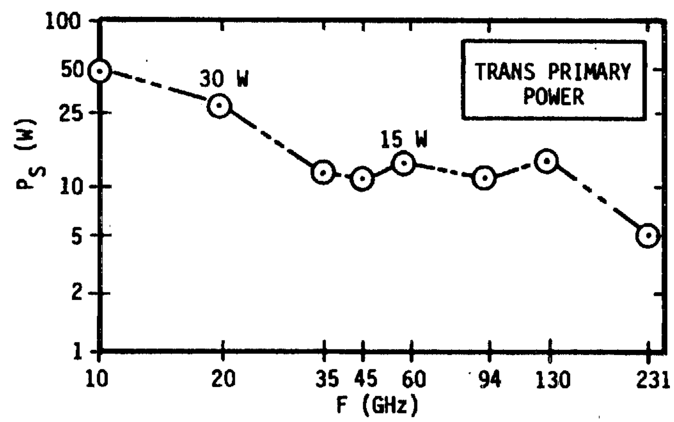
It will be pointed out that there is one advantage to operating at the lower end of the frequency band, a reduced target search time. However, the reduced searched times are purchased at the expense of higher transmitter power and corresponding higher transmitter-receiver isolation requirements.

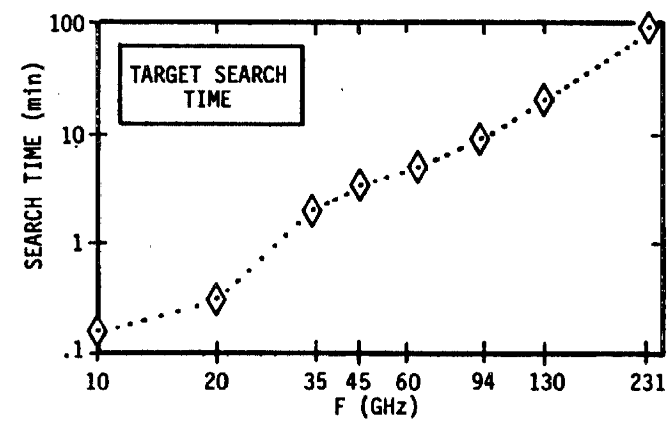
REFERENCE: H. HIESLMAN, C. DESANTIS, AND N. WILSON, "STATE OF THE ART OF SOLID STATE AND TUBE TRANSMITTERS," MICROWAVE JOURNAL, OCTOBER 1983.

FREQUENCY CONSIDERATIONS









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PARAMETERS FOR INITIAL ACQUISITION STUDY

One of the primary considerations in selecting the $\dot{R}/R$ sensor parameters is the requirement to search the 30° by 8 nmi target location uncertainty area. The sensor sensitivity must be adequate to acquire and track a $50M^2$ target at a maximum range of 12 nmi ($1M^2$ target at 4.5 nmi), and the search over the 30° by 8 nmi sector must be completed in 10 minutes. A set of parameters which provides the required performance was established through a series of trade studies.

During the initial acquisition phase the $\dot{R}/R$ sensor will be operating in the Interrupted CW (ICW) mode, transmitting and receiving through a single high-gain gimbaled antenna. For this high duty cycle ICW mode, the transmitter and receiver duty cycles are 40% and 50% respectively and the power delivered to the antenna at the operating frequency of 94 GHz is 500 mW.

Angle coverage is provided by scanning the antenna, starting at the center of the 30° angular sector, and spiraling outward at a constant tangential angular rate. Range coverage is provided by 32 parallel range channels which combined provide an instantaneous coverage of 1.2 km. The required 14.8 km (8 nmi) range search is provided by repositioning the 1.2 km sector; 13 positions provide 8.4 nmi coverage.

The time required to search the 30° by 8 nmi sector was established as a function of the antenna aperture and Doppler resolution. Assuming a noise figure plus losses of 12 dB, the combination of aperture and Doppler resolution was selected to provide the sensitivity required to detect a $50M^2$ target with a 99% detection probability, while minimizing the target search time.

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PARAMETERS FOR INITIAL ACQUISITION STUDY

$R/\dot{R}$ SENSOR PARAMETERS	VALUE
OPERATING FREQUENCY (F_O)	94 GHz
TRANSMITTER POWER (P_T)	500 MW
TRANSMITTER BANDWIDTH (B_N)	4 MHz
NOISE FIGURE PLUS LOSSES (NFL_S)	12 DB
TRANSMITTER DUTY CYCLE (D_T)	40%
RECEIVER DUTY CYCLE (D_R)	50%
RANGE RESOLUTION (ΔR)	37.5M
PARALLEL RANGE CHANNELS (N_{PR})	32
CORRELATOR SAMPLING RATE (F_S)	4 MHz
CORRELATOR EFFICIENCY (η)	-4 DB
SEARCH VOLUME (V_S)	30° X 8 NMI
ANTENNA APERTURE (D)	VARIABLE
DOPPLER RESOLUTION (W_O)	VARIABLE
TARGET PARAMETERS	
RANGE (R_T)	12 NMI
RADAR CROSS SECTION (σ_T)	50 M^2

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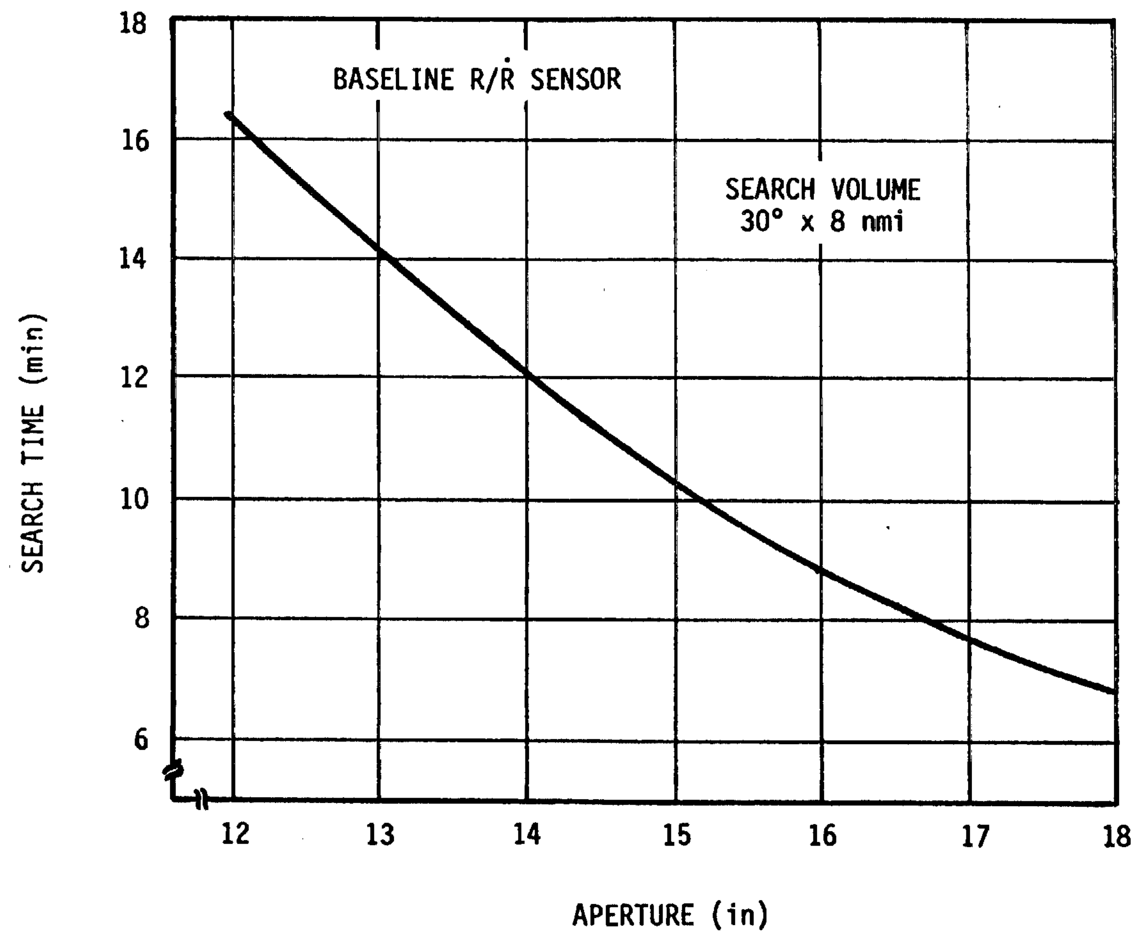
SEARCH TIME VS APERTURE

Results of the initial acquisition study indicate that the time required to search a 30° by 8 nmi sector varies between 16.4 min.and 6.9 min. as the aperture is increased from 12 in. to 16 in. The search time design goal of 10 min. or less is achieved with a aperture slightly larger than 15 in., and selection of a 16 in. aperture for the baseline design would provide the required coverage in less than 9 min.

The search time as a function of aperture was derived by selecting the combination of coherent and incoherent integration times which minimized the search time for that aperture. Selection of a particular value of aperture establishes the signal level (antenna gain) and the number of beam positions (antenna beamwidth) required to provide the 30° angle coverage. As the aperture increases, two counteracting phenomena occur: an increased antenna gain provides an increased SNR while a narrower beamwidth results in an increase in the number of beam positions. Since the number of beam positions is fixed by the aperture, the minimum search time corresponds to the minimum beam dwell time providing the required detection performance. This dwell time is derived by selecting the combination of Doppler and post-detection filter bandwidths (i.e. coherent and incoherent integration times) which minimizes the beam dwell time.

6-177

SEARCH TIME VS APERTURE



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PERFORMANCE COMPUTATION FOR 16 INCH APERTURE

Evaluation of the baseline R/Ṙ sensor performance begins with establishing the SNR available at the correlator output. The factors which determine the output SNR are the input signal power returned from the target, correlator gain, receiver noise power at correlator input, and the transmitter leakage power.

The signal power returned from a $50M^2$ target at the maximum acquisition range of 12 nmi is established by evaluating the Radar Range Equation for the initial acquisition study parameters. The receiver noise power is defined in the transmitted bandwidth of 4 MHz and a receiver noise figure plus losses of 12 dB. The resulting SNR at the correlator input is -22 dB (-118 dBm + 96 dBm).

A correlator gain of 31 dB corresponds to a sampling rate matched to the transmitter bandwidth of 4 MHz and a Doppler bandwidth of 1250 Hz.

The correlator output SNR approaches 9 dB and depends upon the level of transmitter-receiver isolation achieved. Assuming a design goal of 133 dB isolation the corresponding correlator output SNR becomes 8.5 dB.

PERFORMANCE COMPUTATION FOR 16 INCH APERTURE

1. OUTPUT SIGNAL-TO-NOISE RATIO

$$S_O/N_O = \frac{G_C\, S_I}{S_I + N_I + L_I}$$

WHERE S_O/N_O = CORRELATOR OUTPUT SIGNAL-TO-NOISE RATIO
S_I = SIGNAL POWER AT CORRELATOR INPUT
N_I = NOISE POWER AT CORRELATOR INPUT
L_I = TRANSMITTER LEAKAGE POWER AT CORRELATOR INPUT

FOR THE INITIAL ACQUISITION STUDY PARAMETERS,

$$S_I = P_T G_T G_R \lambda^2 \sigma_T D_T^{\;2} / D_R / (4\pi)^3 R_T^{\;4} = -118 \text{ DBM}$$

$$N_I = KTB_N \, NFL_S = -96 \text{ DBM}$$

$$G_C = nF_S/W_O = 31 \text{ DB} \quad \text{WHERE } W_O = 1250 \text{ Hz}$$

S_O/N_O = 9 DB ; $L_I \ll N_I$ (T_X-R_X ISOLATION = ∞ DB)
8.5 DB; L_I = 1/10 N_I (T_X-R_X ISOLATION = 133 DB)
6 DB ; $L_I = N_I$ (T_X-R_X ISOLATION = 123 DB)

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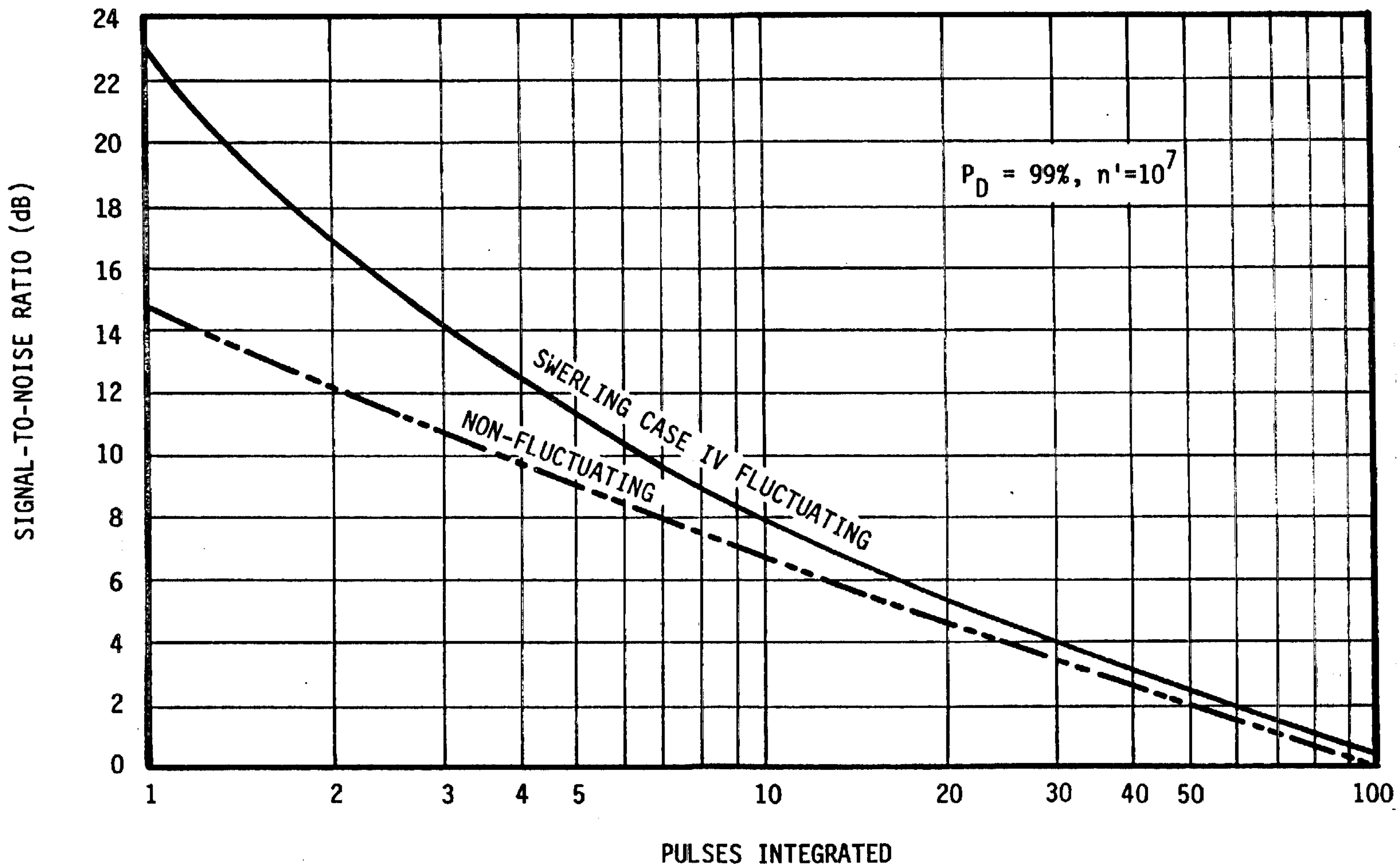
SIGNAL-TO-NOISE RATIO VS NUMBER OF PULSES INTEGRATED

The SNR required to achieve the 99% detection probability goal is dependent upon the target signature characteristics and the target observation time. For the initial acquisition study the Doppler bandwidth of 1250 Hz represents an integration time of 0.8 ms and SNR of 8.5 dB which is not adequate to achieve the desired 99% detection probability. However, by increasing the target observation time through a combination of coherent and incoherent integration, the 99% detection probability goal can be achieved.

Target signature characteristics of the spacecraft targets are not presently available for the RSR waveform and operating frequency of 94 GHz, therefore, an assumption will be made concerning the target model. A Swerling Case IV model will be assumed to represent a satellite target which might be represented as one large reflector together with a number of smaller reflectors and a signature which is subject to small changes in orientation. It is anticipated that during a single integration period of 0.8 ms the target signal strength will remain constant; however, target signature fluctuations will occur from one integration period to the next. Radar signature measurements of complex targets such as armored vehicles using a Bendix developed 94 GHz instrumentation radar, substantiate the sensitivity of radar signature to change in range and aspect angle. This is not surprising when considering the fact that a complex target is comprised of many reflective surfaces which may be large with respect to the 3.2 mm wavelength of the transmitted energy.

In order to achieve a 99% detection probability for the Swerling Case IV fluctuating target it will be necessary to incoherently integrate for 7.2 ms which corresponds to 9 pulse periods. A pulse period in this context represents the correlator coherent integration period of 0.8 ms.

SIGNAL-TO-NOISE RATIO VS NUMBER OF PULSES INTEGRATED



PERFORMANCE COMPUTATION FOR 16 INCH APERTURE (CONTINUED)

The 7.2 ms target observation period achieved through a combination of coherent and incoherent integration represents the time that the beam dwells during the angle-range search.

Angle search is performed with a spiral scan which starts at the center of the 30° angular sector and spirals outward at a constant tangential angular rate of 54 deg/s. The number of beam positions covered during the angle search (5659) is a function of the angular coverage (30°), two-way beamwidth (0.39°) and the beam overlap factor (20%). The time required to search the 30° sector is equal to the number of beam positions times the dwell time at each beam position: 5659 x 7.2 ms = 0.68 min.

An instantaneous range coverage of 0.65 nmi is provided by 32 parallel range bins therefore 13 positions of the 0.65 nmi sector are required to provide coverage of the 8 nmi target location range uncertainty. Since a 30° angle search is performed at each of the 13 range gate positions the total search time is 8.8 min.

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PERFORMANCE COMPUTATION FOR 16 INCH APERTURE (CONTINUED)

2. DETECTION PROBABILITY

FOR A SWERLING CASE IV FLUCTUATING TARGET, INTEGRATION FOR 9 PULSE PERIODS RESULTS IN A DETECTION PROBABILITY OF 99%. THIS REQUIRES A BEAM DWELL TIME OF 7.2 MS (9 X 1/1250 Hz).

3. ANGLE SEARCH TIME

THE TIME REQUIRED TO SEARCH A 30° INCLUDED ANGLE CONICAL SECTOR, USING A 20% OVERLAPPING SPIRAL SCAN,

$$T_S = \left(\frac{\theta_C}{2\theta_B} - \frac{1}{2}\right)^2 \frac{\pi}{(1-\theta_O/\theta_B)} T_D = \left(\frac{30}{2 \times 0.39} - \frac{1}{2}\right)^2 \frac{\pi}{(1-0.2)} \; 7.2 \text{ MS} = 0.68 \text{ MIN}$$

4. ANGLE-RANGE SEARCH TIME

THE 32 PARALLEL RANGE BINS COVER 1200M (0.65 NMI), CONSEQUENTLY 13 POSITIONS PROVIDE 8.4 NMI RANGE COVERAGE AND THE TIME REQUIRED TO PROVIDE 30° BY 8 NMI COVERAGE,

$$T_T = 13 \times 0.68 \text{ MIN} = 8.8 \text{ MIN}$$

PARAMETERS FOR CW MODE STUDY

Following the initial search to locate the target and subsequent acquisition, the R/$\dot{R}$ sensor tracks the spacecraft target providing the OMV with angle, range and range rate data. During this period of time the R/$\dot{R}$ sensor operates in the Interrupted CW mode, transmitting and receiving through a single high-gain antenna. At a range of 500 ft. the sensor mode of operation is changed from ICW to a higher accuracy CW mode which utilizes separate antennas for transmission and reception. As the OMV continues to approach the target the R/$\dot{R}$ sensor provides accurate measurements of range and range rate.

In order to achieve the range accuracy goal of 6 in, the range resolutions in the CW mode is increased to 1.2M. This requires increasing the transmitted bandwidth to 125 MHz. Correspondingly the correlator sampling rate is increased to 62.5 MHz and the correlator gain increases to 43 dB.

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Results of a study using these CW mode parameters indicate that the transmitter power and corresponding isolation required for operation in the CW mode is a function of the angle coverage provided. Signal detection and measurement accuracy computations were based on a target radar cross section of $1M^2$ located at a range of 500 ft. The maximum transmitter power of 500 mW provides required performance for a 30° angle coverage.

PARAMETERS FOR CW MODE STUDY

$R/\dot{R}$ SENSOR PARAMETERS	VALUE
OPERATING FREQUENCY (F_O)	94 GHz
TRANSMITTER POWER (P_T)	500MW (MAX)
TRANSMITTER BANDWIDTH (B_N)	125 MHz
RANGE RESOLUTION (ΔR)	1.2 M
DOPPLER RESOLUTION (W_O)	1250 Hz
CORRELATOR SAMPLING RATE (F_S)	62.5 MHz
CORRELATOR GAIN (G_C)	43 DB
NOISE FIGURE PLUS LOSSES (NFL_S)	12 DB
ANTENNA APERTURE (D)*	VARIABLE

TARGET PARAMETERS	
RANGE (R_T)	500 FT.
RADAR CROSS SECTION (σ_T)	1 M^2

*IDENTIAL TRANSMITTER AND RECEIVER ANTENNAS ASSUMED

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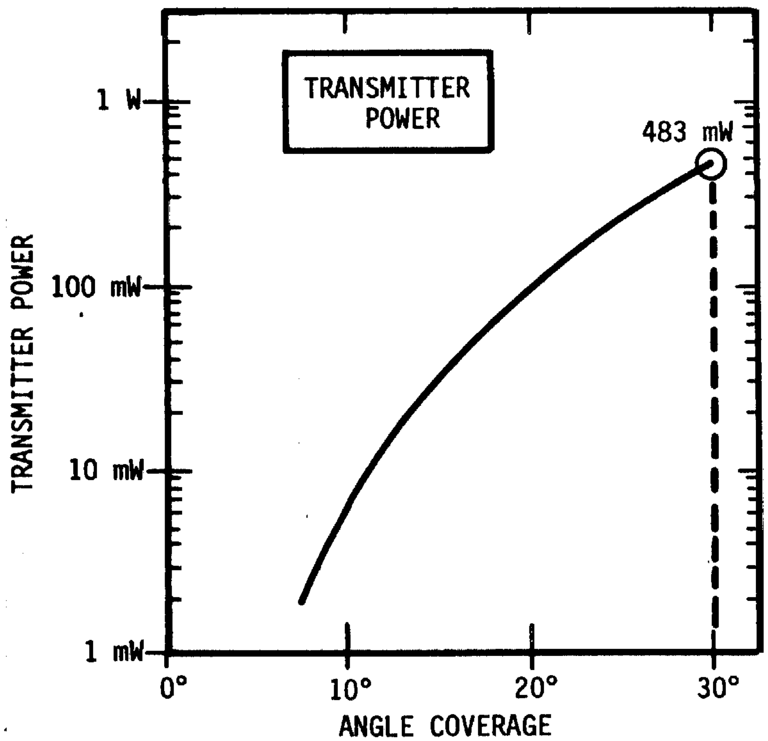
CW MODE TRANSMITTER POWER AND ISOLATION REQUIREMENTS

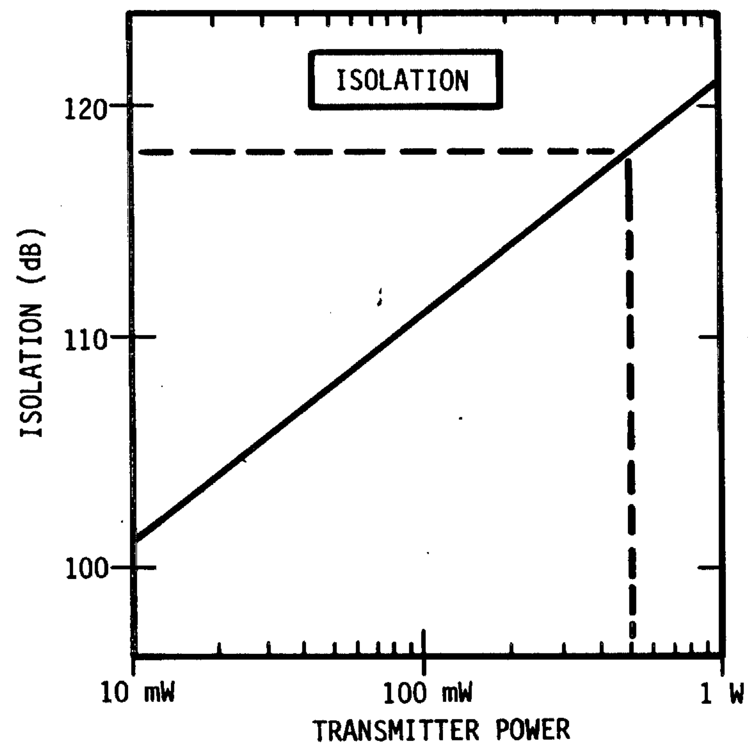
The transmitter power required for CW operation is a function of the angle coverage provided and varies between 483 mW to provide 30° coverage to less than 10 mW for 10° coverage. Since the transmitter and receiver antennas are body fixed, a 30° angle coverage would be compatible with the coverage provided by the ICW mode gimbaled antenna.

However, if the angle coverage required during the last 500 ft. were less than 30°, a lower powered transmitter could be used. For instance a 95 mW transmitter would provide 20° coverage, and if this were implemented as a separate transmitter, a degree of redundancy would be provided.

The isolation required in the CW mode of operation is a direct function of the transmitted power. An isolation between transmitter and receiver of 118 dB is required for the maximum transmitted power level of 500 mW. Achieving this level of isolation between antennas which are located relatively close together appears to be feasible at the millimeter wave operating frequencies of the R/R sensor.

CW MODE TRANSMITTER POWER AND ISOLATION REQUIREMENTS





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OMV RANGE/RANGE RATE SENSOR BLOCK DIAGRAM

The Range/Range Rate sensor is divided into four major components related to functions performed: Transmitter/Receiver, Angle Processor, Range/Doppler Processor, and Data Processor.

The basic function of the transmitter is to generate the random waveform required for target acquisition and tracking. A random code is generated by sampling random noise and the resulting binary code is used to biphase modulate an S-Band source. The S-Band modulated signal is then upconverted to the W-Band operating frequency and amplified. The transmitted PRF is selected to provide a peak response at the target range, and is adjusted as the range changes in order to maintain target visibility. A programmed attenuator provides automatic power control and prevents receiver saturation as the OMV approaches the target satellite.

The basic functions of the receiver are to translate the target return signal RF spectrum to a video spectrum, and to generate the signals required for angle, range, and doppler processing. A monopulse arithmetic network converts the four quadrant antenna outputs into sum (Σ) and difference (Δ) functions which are translated to video through a double conversion process.

The angle processor converts the Σ and Δ amplitude functions into equivalent phase functions ($\Sigma + \Delta$ and $\Sigma - \Delta$) and generates the appropriate angle tracking control signals for phase monopulse processing.

The range tracking loop is implemented using an early-late gate range tracking algorithm. Doppler tracking is implemented as a conventional AFC loop using a frequency discriminator and voltage controlled oscillator.

The data processor provides all the sensor control functions required for target search and track. Transmitter control functions include noise bandwidth, pulse repetition rate, attenuation control operating mode control, and delay line positioning. Each of the tracking loops is closed through the data processor.

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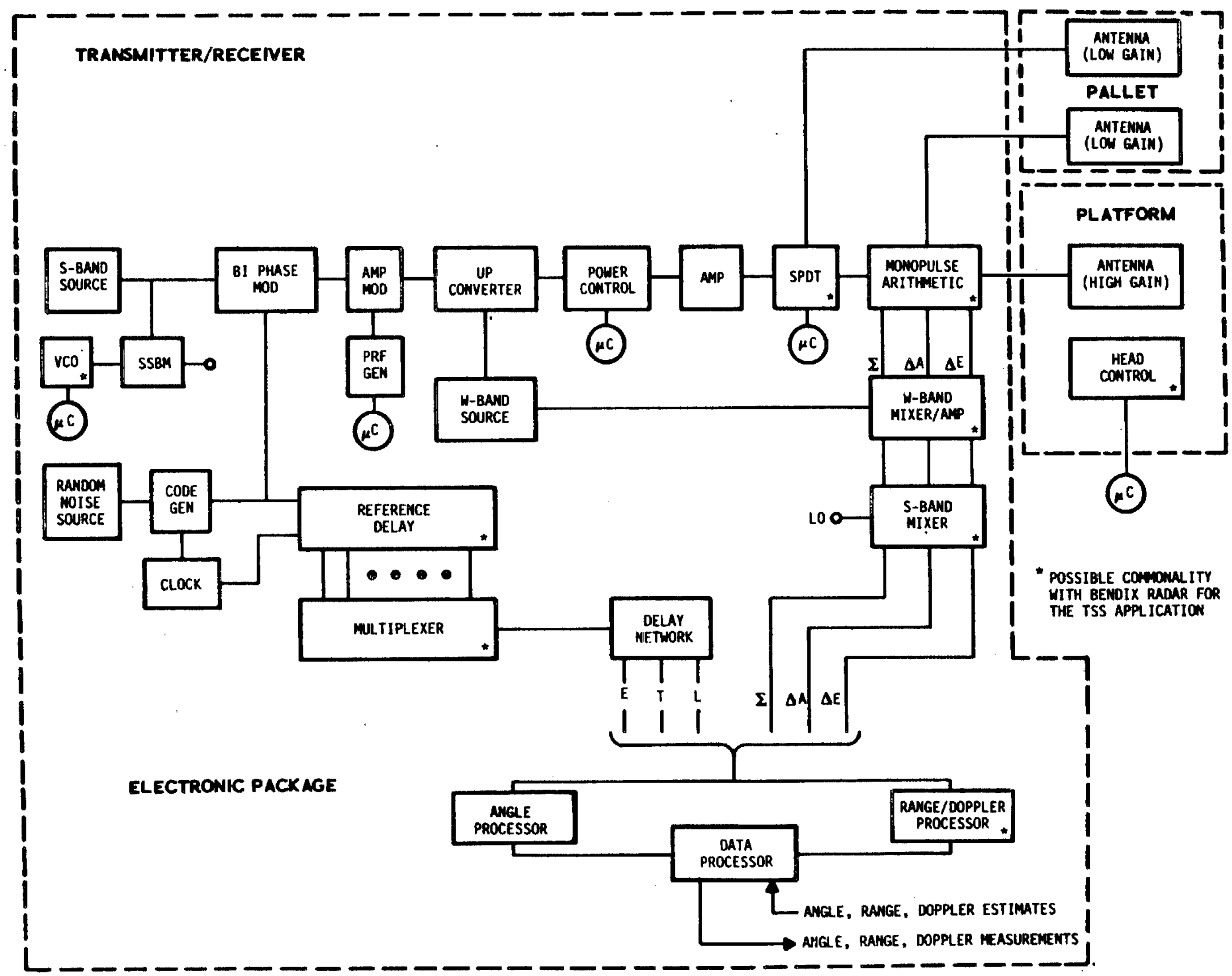
OMV RANGE/RANGE RATE SENSOR BLOCK DIAGRAM



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RANGE/RANGE RATE SENSOR DEVELOPMENT PROGRAM

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RANGE/RANGE RATE SENSOR DEVELOPMENT PROGRAM

The baseline Range/Range Rate sensor described in the previous sections is a combination of concepts, some of which have been demonstrated with existing RSR brassboards, and others that are yet to be demonstrated. A program plan has been formulated which has the objective of developing the R/R sensor through a series of steps which will expand present RSR capabilities to satisfy OMV application requirements.

The first step in the program plan is the development of a set of sensor requirements. Such requirements as initial acquisition search volume and search time, tracking and measurement accuracies are to be specified.

Using this set of requirements as system inputs, a baseline R/R sensor configuration is then defined to satisfy these requirements. In this second program step a series of trade studies will be utilized to establish sensor parameters such as operating frequency, transmitter power, antenna aperture and processing gain.

Having established the baseline configuration, the third program step addresses the major sensor functional components. The Transmitter/Receiver, the Angle, Range, and Range Rate (Doppler) Tracking Loops, and accompanying Data Processor are designed to satisfy accuracy and response requirements.

In addition to analytical studies, hardware testing will be utilized to evaluate design concepts. Existing RSR brassboards will be appropriately modified to the baseline configuration and performance testing will be used to evaluate design concepts.

The series of steps outlined is viewed as an ongoing process in which activities in all areas would be occurring simultaneously.

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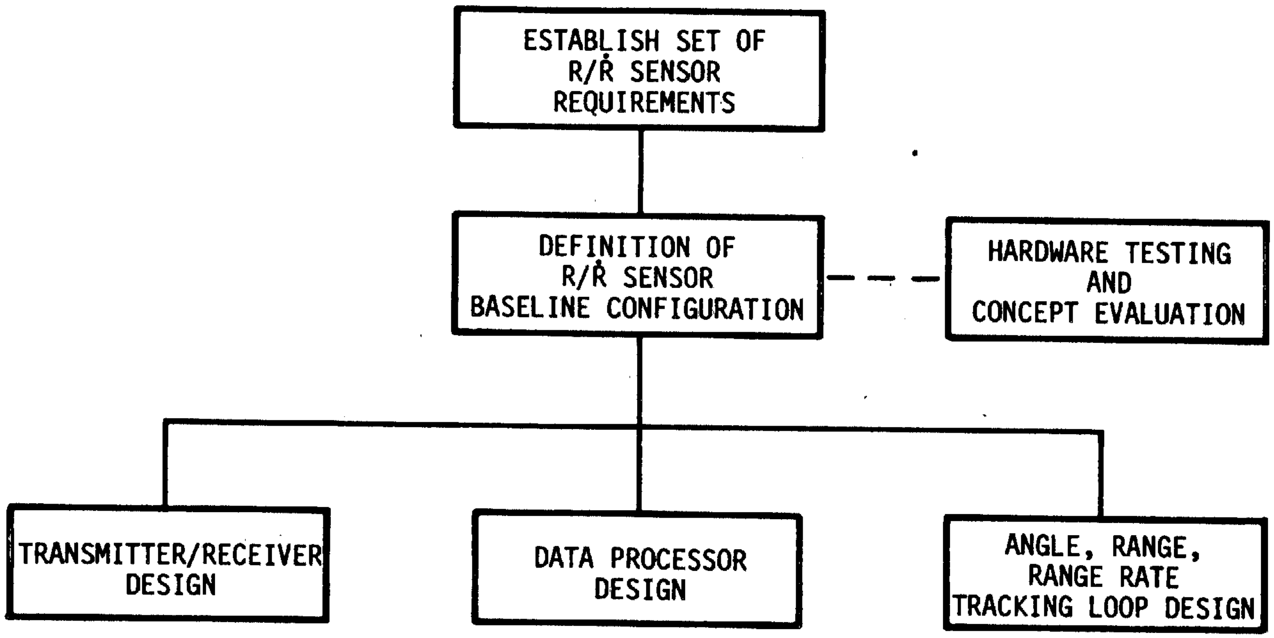
RANGE/RANGE RATE SENSOR DEVELOPMENT PROGRAM



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RANGE/RANGE RATE SENSOR PROGRAM PROGRESS

During the past two years Bendix has been working with NASA MSFC and OMV contractors on the OMV R/Ṙ sensor conceptual design. As a result of this effort a preliminary set of R/Ṙ sensor goals have been established, and a baseline sensor conceptual design has been completed with parameters selected to satisfy those goals.

The sensor design combines mature and developing technologies. The mature technology represents 13 years experience in development of the Random Signal Radar which has resulted in 4.2 GHz and 10 GHz radar altimeters, a 10 GHz monopulse seeker, and a CW millimeter wave instrumentation radar and high resolution millimeter wave seeker both operating at 94 GHz.

The developing technology combines CW and Interrupted CW modes of operation, an increased receiver sensitivity, and target tracking capability to essentially zero range, with the performance required to accomplish successful docking of two spacecraft.

A hardware test program is presently being conducted which has the objective of developing the technologies required to realize the R/Ṙ sensor. Through a combination of analysis, modifications to existing RSR millimeter wave brassboards, and testing, these developing technologies will be transformed into a low risk category.

In addition to the hardware test program, consideration has been given to other areas which require further investigation. Areas of concern include target radar signature characteristics, near field operation, and implications of sensor location on sensor requirements. These areas are to be addressed as the program progresses and evaluated in terms of their impact on sensor design.

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RANGE/RANGE RATE SENSOR PROGRAM PROGRESS

- PRELIMINARY SET OF R/$\dot{R}$ SENSOR PARAMETERS ESTABLISHED.
- BASELINE SENSOR CONCEPTUAL DESIGN COMPLETED.
- HARDWARE TEST PROGRAM IN PROGRESS.
- AREAS OF FURTHER INVESTIGATION PROPOSED.

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HIGH RESOLUTION MMW SEEKER

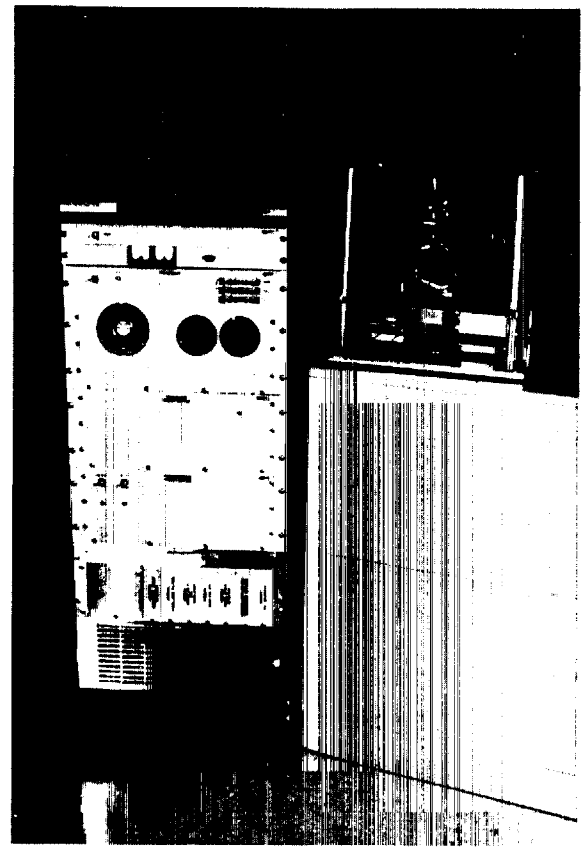
$R/\dot{R}$ SENSOR TEST HARDWARE-HIGH RESOLUTION MMW SEEKER

One of the Random Signal Radars being used in the Hardware Test Program for development of the $R/\dot{R}$ sensor is the High Resolution Millimeter Wave Seeker. The monostatic radar which operates in a high duty cycle interrupted CW mode, transmits a random waveform centered at the operating frequency of 94 GHz. Separate angle, range, and Doppler tracking loops are incorporated to maintain track of the target following the initial search and acquisition phase.

The gimbaled seeker head is packaged in a spherical shell which contains all the millimeter wave and microwave components. The correlator and data processing circuitry with accompanying power supplies are located in an equipment rack.

The High Resolution Seeker will be used in the development of the ICW mode of operation.

R/Ṙ SENSOR



TEST HARDWARE
HIGH RESOLUTION MMW SEEKER

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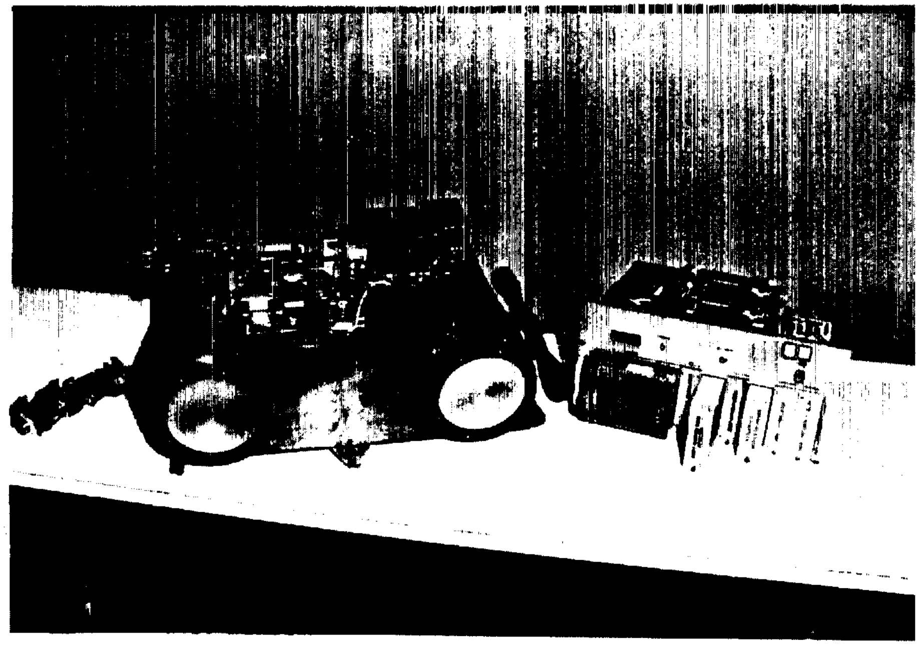
MMW INSTRUMENTATION RADAR

R/Ṙ SENSOR TEST HARDWARE-MMW INSTRUMENTATION RADAR

The MMW Instrumentation Radar is likewise being utilized in the Hardware Test Program in development of the CW mode of operation. The CW radar which operates at 94 GHz will be used in the evaluation of isolation achievable between transmitter and receiver paths. Antenna pairs of 1.5°, 3°, 7° and 22° beamwidths will be used to establish the isolation as a function of antenna beamwidths.

$R/\dot{R}$ SENSOR



TEST HARDWARE
INSTRUMENTATION RADAR

PROPOSED R/$\dot{R}$ SENSOR DEVELOPMENT ACTIVITY

Several areas of investigation related to the development of the R/$\dot{R}$ sensor have been identified. The activities are designed to provide the additional information required for further definition of R/$\dot{R}$ sensor parameters.

Establishment of the angle, range, and range rate functions of time is essential in defining initial acquisition search parameters. The coverage required and the time available to provide the coverage are key design parameters as was illustrated previously. Consideration should be given to rendezvous maneuvers which provide search times compatible with a solid-state implementation and minimization of size, weight, and power requirements.

Location of the sensor influences the angle coverage provided in the close-in operation as well as the utility after docking has been accomplished. A combination of a boom mounted antenna and an antenna location near the docking probe will provide angle coverage from initial acquisition to target docking and continue to provide coverage of a secondary target.

It is anticipated that the target will have an extended signature characteristic when the R/$\dot{R}$ sensor is operating in the high resolution mode. Measurements of the target signature at the operating frequency of 94 GHz are required for development of extended target tracking algorithms.

During the final phases of docking the target will be in the near field of the R/$\dot{R}$ sensor antennas. Since antenna pattern characteristics in the near field may degrade sensor tracking performance it is necessary to establish the tracking characteristics for near field operation.

Other areas of development activity include evaluation of the operational environment on performance and generation of a computer simulation to aid in sensor design and performance evaluation.

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PROPOSED R/$\dot{R}$ SENSOR DEVELOPMENT ACTIVITY

- EXPANSION OF SENSOR REQUIREMENTS DEFINING ANGLE, RANGE, AND RANGE RATE FUNCTION TIME HISTORIES.
- IMPLICATIONS OF SENSOR LOCATION ON REQUIRED SEARCH AND TRACKING FUNCTIONS.
- MEASUREMENT OF TARGET RADAR SIGNATURE CHARACTERISTICS.
- INVESTIGATION OF NEAR FIELD OPERATIONAL CHARACTERISTICS.
- EVALUATION OF OPERATIONAL ENVIRONMENT ON R/$\dot{R}$ SENSOR PERFORMANCE.
- GENERATION OF COMPUTER SIMULATION TO AID IN SENSOR DESIGN AND PERFORMANCE EVALUATION.

RENDEZVOUS & PROXIMITY SENSOR CANDIDATES

B. Kunkel

MBB-ERNO

Ottobrunn, F.R.G.

RENDEZVOUS & PROXIMITY OPERATIONS WORKSHOP

Lyndon B. Johnson Space Center

Houston, Texas

February 19-22, 1985

RENDEZVOUS & PROXIMITY SENSORS

RVD + PROXIMITY SENSORS
INTRODUCTORY REMARKS

- Emphasis on Rendezvous/Docking (RVD) mission tasks until now
- Proximity RVD, however, reveal great commonality with proximity operations (in-orbit servicing, replacement/repair, exchange, re-fuel etc.)
- Optical (laser, imaging sensors) sensors conceived with multiple optical heads to provide post-RVD usage (e.g. operations control/monitoring)
- All sensors conceived as autonomous navigation sensors, directly interfacing with and guiding the AOCS of the approaching S/C

Rationale: even with presence of astronauts and full visual control capability, man is considered too inaccurate, too slow and too subtle to safely conduct such critical manoeuvres as RVD with a tangeable space station or other modules; also, autonomous concept enables RVD with unmanned platforms as well

- All sensors investigated until now have been verified in terms of technology, applicability and, too a great extent, performance

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RENDEZVOUS & PROXIMITY SENSORS

RVD REQUIREMENTS WITHIN COLUMBUS MISSION SCENARIO
(RVD Sensor Related, expected Requirements)

RVD MISSION TASKS:

- Navigation support towards target module, providing:
  - Target position coordinates ($\leq 0.1^\circ$ in near-field)
  - Range (resolution $\leq 1‰$ of range), range rate (relative velocity - few mm/s in near field)
  - Angular rates between S/C's inertial references (about 0.1° /s assumed)
- Relative attitude measurement + control between approaching COL. MODULES (support onboard AOCS + ground/in-orbit control), i.e. bearing angles, pitch + yaw (resolution 0.1° resp. 1° ass.)
- Ranges to be covered
  15 km - 0 m; at longer distances, guidance is expected to be provided by ground stations
- Autonomy of sensors is assumed to be advantageous (to also cover unmanned module approaches)

RENDEZVOUS & PROXIMITY SENSORS

RVD SENSOR MEASUREMENT REQUIREMENTS ASSUMPTIONS

RENDEZVOUS PHASE	AT RANGES	RANGE	RANGE RATE	LATERAL POSITION	LATERAL RATE	RELATIVE ATTITUDE	ATTITUDE RATE
Homing	beyond 10km 10-1km	500m 50m	1m/s 1m/s	-- --	0.1m/s 0.1m/s	-- --	-- --
Final Approach	1km-100m 100m-10m	10m 0.5m	0.2m/s 0.1m/s	5m 0.25m	0.1m/s 0.05m/s	-- --	-- --
Terminal Closure	10m-1m	--	0.05m/s	0.10m	0.005m/s	2°	--
Docking: Mechanically Controlled:	1m	--	0.05m/s	0.10m	0.005m/s	< 2°	< 0.1°/s
AOCS Controlled	0m	--	0.005m/s	0.02m (≤0.001m?)	$\boxed{0.0005m/s}$	< 0.5°	< 0.05°/s

$\boxed{}$ considered critical

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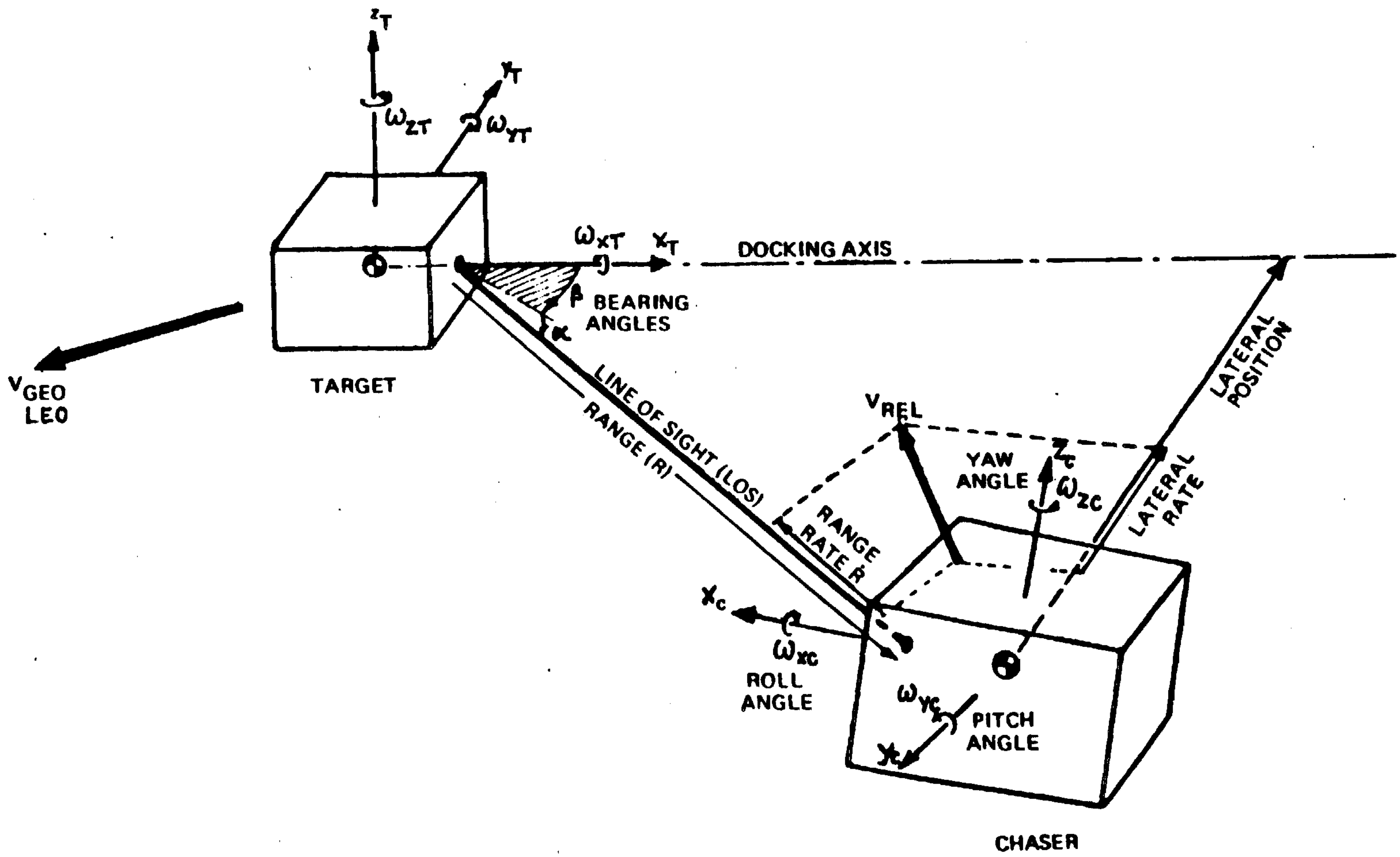
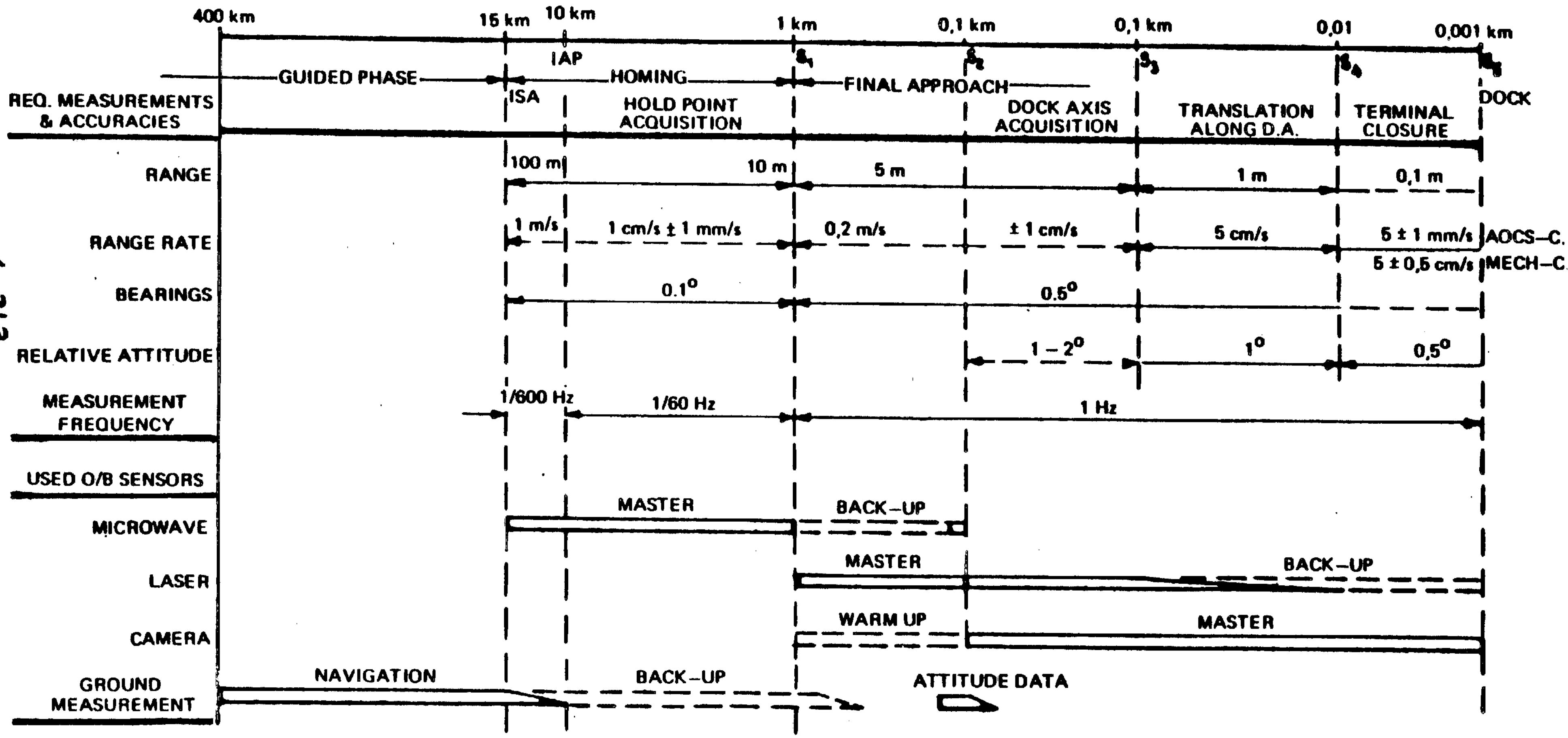
RENDEZVOUS & PROXIMITY SENSORS

DEFINITION OF REFERENCE SYSTEM AND MEASUREMENT PARAMETERS



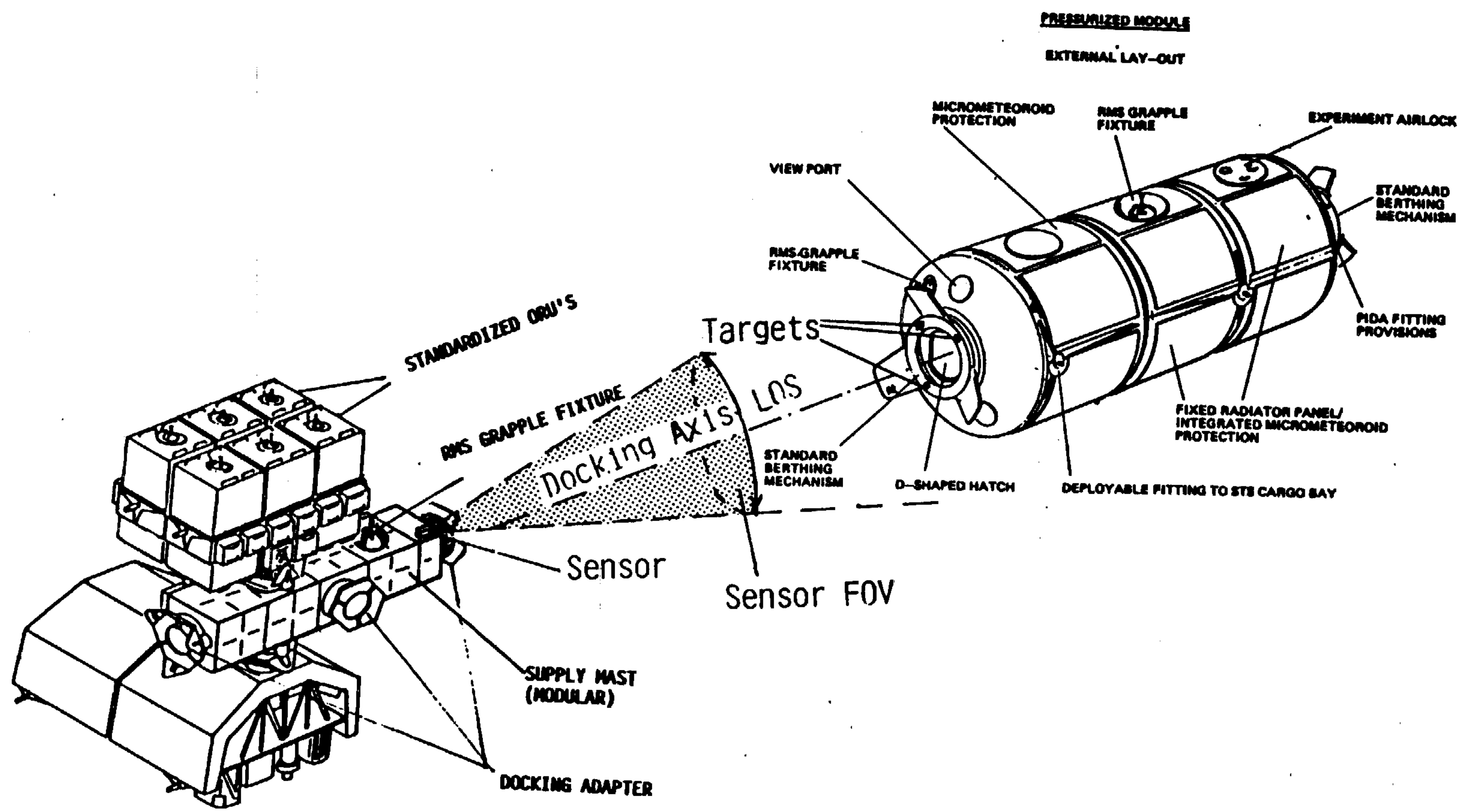
RENDEZVOUS & PROXIMITY SENSORS

MAIN RVD MISSION PHASES
(Example: GEO RVD, unmanned platforms)



RENDEZVOUS & PROXIMITY SENSORS

EXAMPLE OF POSSIBLE RVD APPROACH BETWEEN SPACE STATION ELEMENTS : COLUMBUS WITH PAYLOAD CARRIER



RENDEZVOUS & PROXIMITY SENSORS

SURVEY OF INVESTIGATED SENSORS (all "SMART" SENSORS)
(Studies and experimental verification)

SENSOR TYPE	ROLE IN RVD				RANGE DOMAIN (m)	REMARKS
	R	RR	RA	AR		
Microwave Sensor -S-band Doppler Radar	X	X	(X)	(X)	$20x10^3$-100	- Sensitive to/producing EMI - Poor near-field performance - Active target on target S/C needed
-K_u-band PRN Range Finder	X	X	X	(X)	$20x10^3$ 10	- EMI as above - Better near-field performance - Omni-direct. mode feasible
Laser Diode Radar Sensor (cw/pulsed)	XX	XX	X	X	$20x10^3$-0.1	- phase detection or TOF measurement - Omni-direct. beam deflection - Direct R,RR, angular position information
Imaging Sensor (Solid-state cameras + similar)	X	X	XX	XX	100-0.3	- Highest processing effort for R,RR,AR - Active targets (LEDs,...) on target S/C needed Fix-focus/limited range or variable optics
Imaging Sensor + Laser as Illum	X	X	XX	XX	200-0.3	

R : Range, resolution; RR: Range Rate; RA: Relative Attitude (angles); AR: Angular Rates

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RENDEZVOUS & PROXIMITY SENSORS

SENSOR CANDIDATES SURVEY AND RANGE COVERAGE
(sequence according to capabilities/priority)

1. <u>Long Range</u> ($\geqslant$ 10km $\rightarrow$ 1km):

   - MW sensors (S- or K_u-band Radar)
   - Laser Diode optical radars (cw and pulsed)
   - Solid-state cameras with long focal length

2. <u>Medium Range + Transfer Phase</u> (1km $\rightarrow$ 0.1km):

   - Laser radars
   - MW sensors
   - Solid-state imaging sensors (distinct target tracking + contour tracking)

   Note: Transfer Phase requires large azimuth + elevation tracking FOV!

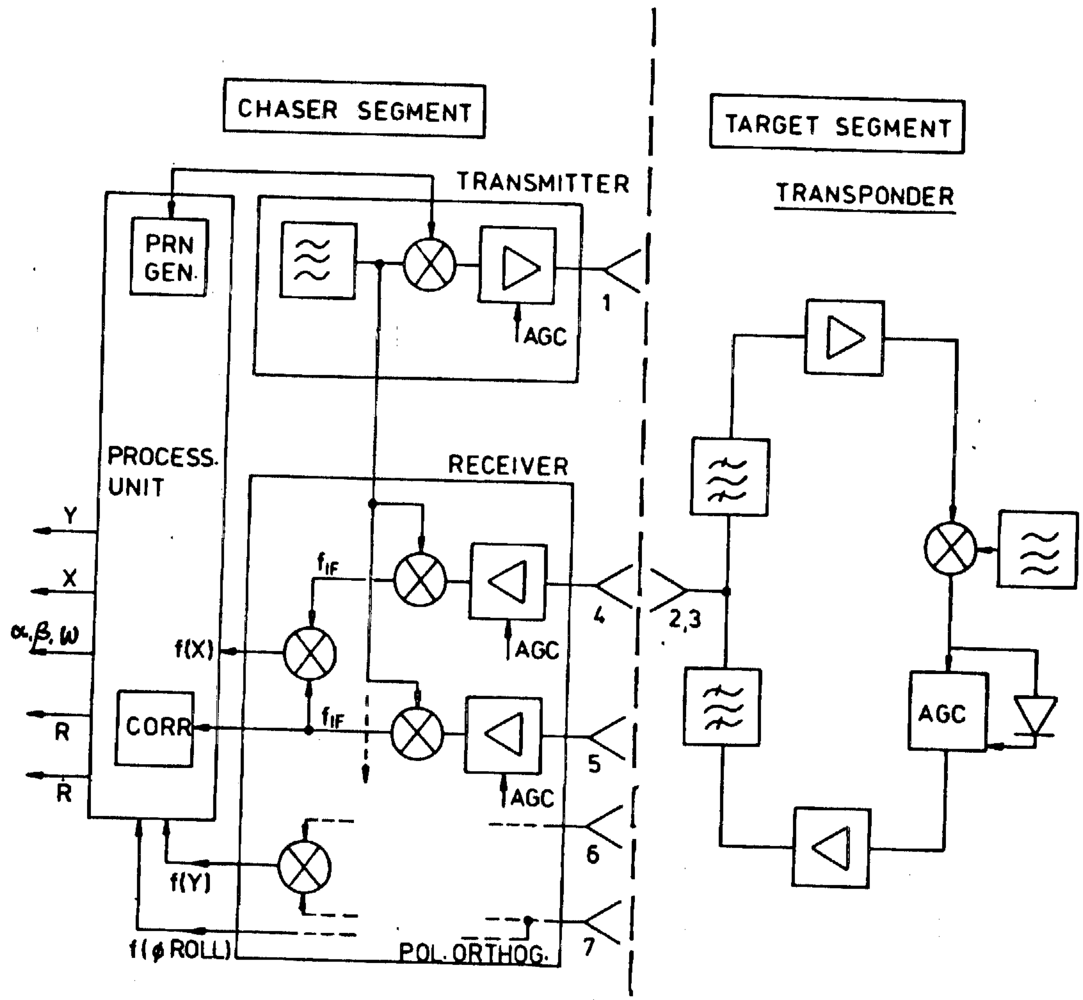
3. <u>Near-field to Proximity</u> (100m $\rightarrow$ 0.1m, Docking axes in LOS):

   - Solid-state imaging sensors
   - Laser sensors

RENDEZVOUS & PROXIMITY SENSORS

PRN RADAR FUNCTIONAL DIAGRAMME

(Preferred Concept)



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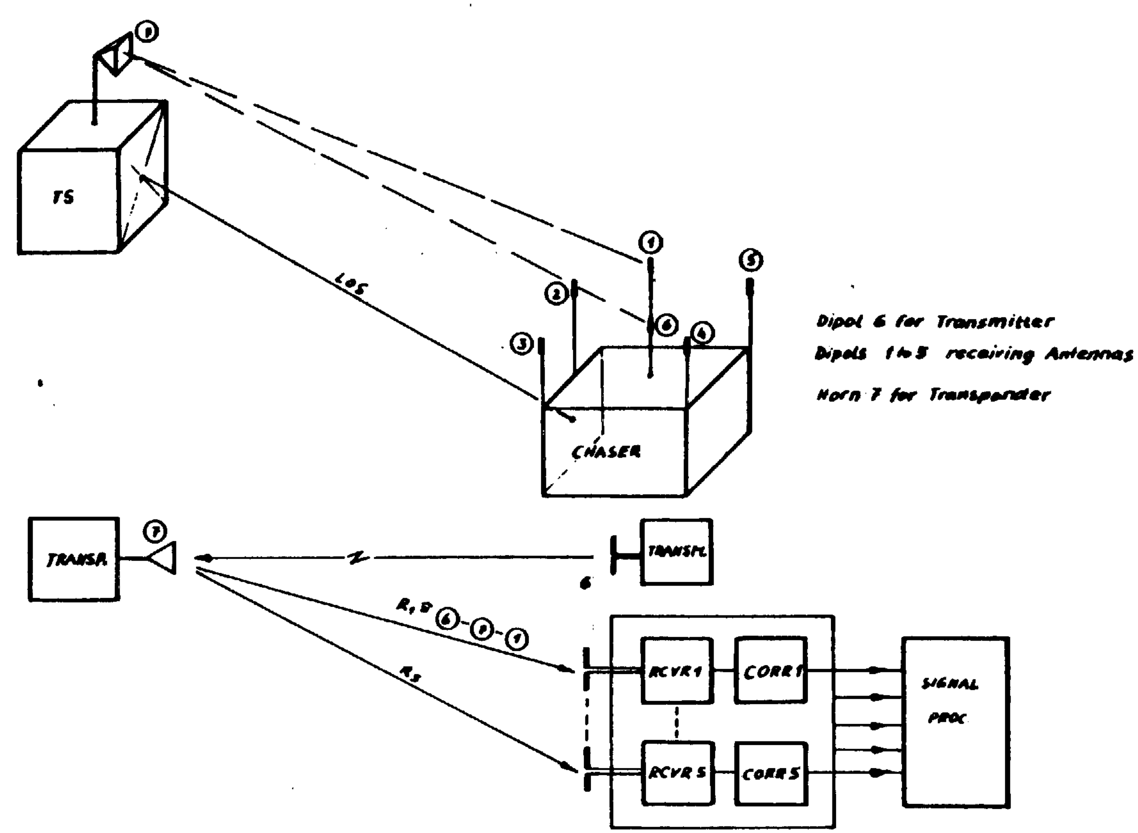
REVISED WIDE-ANGLE PRN INTERFEROMETRIC TRACKING RADAR ARRANGEMENT



RENDEZVOUS & PROXIMITY SENSORS

PRN RADAR SUMMARY PERFORMANCE

	Previous PRN K_u-band Radar	Revised wide-angle version
<u>Ranging:</u>		
● Range Resolution (m)		
- 10 km	0.1	0.1
- 100 m	0.05	0.05
- 10 m	0.01	0.01 with add. transp. on dock. panel
● Range Rate Resol. (m/s)		
- 10 km	0.1	same
- 100 m	0.05	
- 10 m	$\leq$ 0.01	
<u>Angular Position Determination</u> ($\leq$ 100 m, deg)		
Angular Resolution:		
Bearing Angles	0.1	$\approx$ 1
Roll Angle	0.2	$\approx$ 1
Pitch, Yaw	$\pm 3 \div 1$ ($\geq$ 10m)	$\pm 5 \div \pm 2$ ($\geq$ 10m)
Angular Rates (°/s)	$\approx$ factor of 5 better than above values through statistics	
FOV Capability (deg)	5 x 100 (100 in elevation)	360° (azimuth) x 50° (elevation)
Frame Time (s)	$\leq$ 0.01	$\leq$ 0.01

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RENDEZVOUS & PROXIMITY SENSORS

MAIN K_u-BAND PRN RANGING AND TRACKING SENSOR

SPECIFICATIONS AND INTERFACE REQUIREMENTS

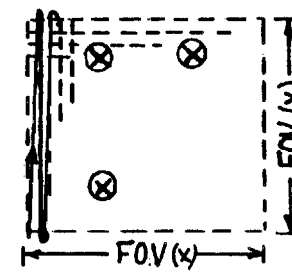
	Nominal Sensor	Revised omni-dir. Sensor
● Transceiver at chaser:	5 antennas	6 antennas
- Mass:	2.8 kg	3.6 kg
- Volume:	10x10x10cm without antennas	same
- Power consumption:	4 W	5 W
- Transmitter power	100 mW	250 mW
● Electronics at chaser:		
- Mass:	0.8 kg	1.6 kg
- Volume:	$10x10x10cm^3$	$10x15x10cm^3$
- Power cunsumption:	2 W	3.2 W
● Transponder at target S/C:	1 antenna	same
- Mass:	1.2 kg	same
- Volume:	10x10x7cm	same
- Power consumption:	2 W	same
- Emitted power	100 mW	same
● Interface Requirements		
- free FOV:	30° x 50°	± 25° x 360° (antennas erected)
- Data Rate:	t.b.d.	t.b.d.
- EMC	"	"
- Thermal Contr./oper. temper.	- 20/+50° (electr.)	same

RENDEZVOUS & PROXIMITY SENSORS

OPTICAL RVD SENSORS -
LASER DIODE SYSTEMS INVESTIGATED

(a) CONTINUOUS WAVE (CW) LASER SENSORS:

- Using phase difference between outgoing modulation frequency and reference signal as direct measure of range, range rate
- Angular information by narrow-beam scanning of a given FOV resp. L-shape 3-targets configuration
- Requires 2 modulation frequencies for longer ranges to cope with range ambiguity (of modulation frequency)



(b) PULSED LASER DIODE SENSORS:

- Using time-of-flight measurement (ultra-fast electronics) as direct measure of range, range rate
- Angular information as (a)
- Covering all ranges with same electronics
- Higher beam energy density than cw laser, thus shorter scan times (dwell times on target) feasible

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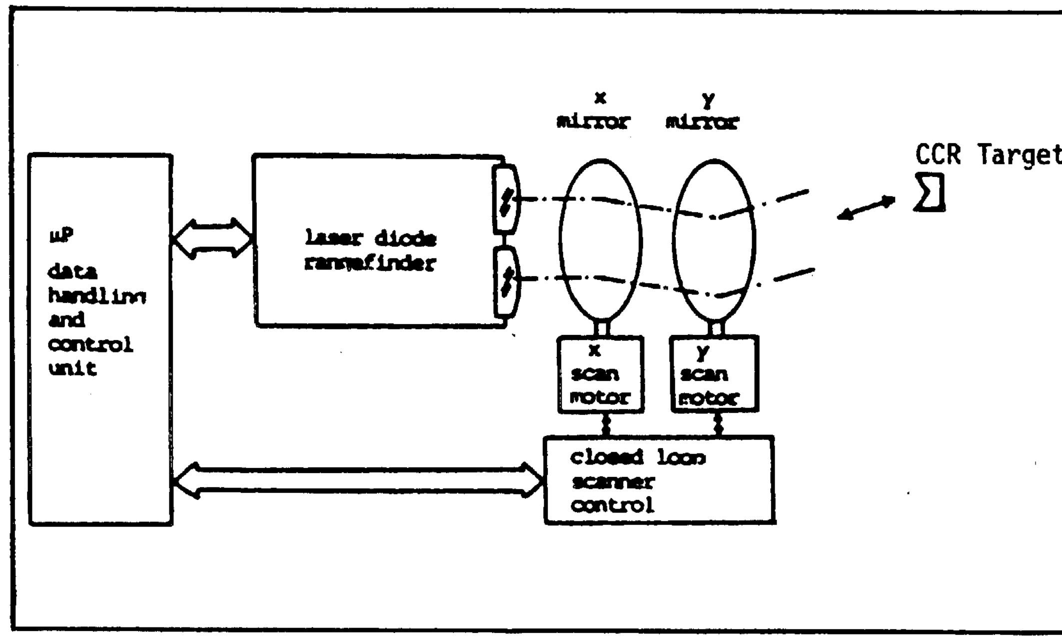
RENDEZVOUS & PROXIMITY SENSORS

LASER DIODE RANGE + ATTITUDE MEASUREMENT SENSORS



MAIN SUBUNITS (see Figure):

- Laser Diode Transmitter (Diode, Optics)
- Retro-Reflector Target(s)-(CCR)
- Receiver (PAD + Electronics)
- Two-axis Beam Deflection
- Data Handling/Processing + Control Electronics

MAIN ADVANTAGES:

- High range resolution for distances of concern, growth potential
- High bearing/roll angle resolution (if there is time for interpolation, below beam divergence of about 0.5 mrad or 0.029°)
- Passive reflector targets on TS only, no need for a command link to the TS
- Compact sensor
- Lowest processing effort (range information is generated directly as well as target coordinates)
- Functional models already established (except pitch/yaw extraction)

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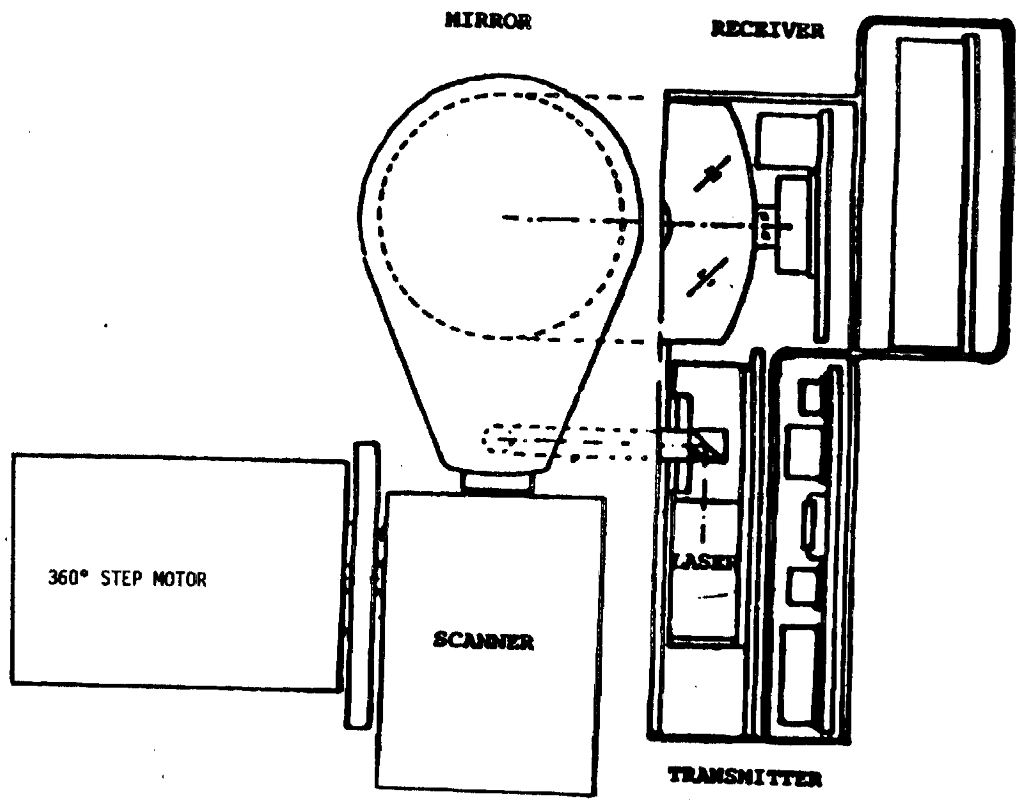
RENDEZVOUS & PROXIMITY SENSORS

EXAMPLE OF COMPACT CW LASER SENSOR HEAD WITH
360° ROTATION MOUNT (AZIMUTH)



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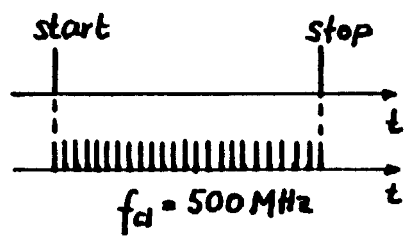
PULSED LASER DIODE RANGEFINDER

PRINCIPLE OF OPERATION

TIME INTERVAL MEASUREMENT: TIME-TO-DIGITAL CONVERSION



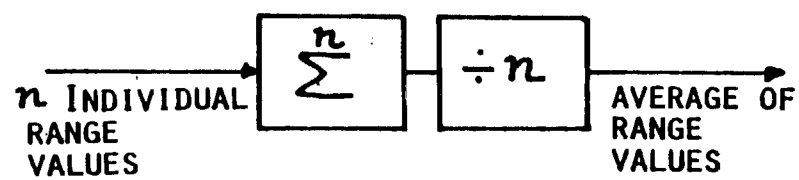
THE TIME INTERVAL BETWEEN "START" AND "STOP" SIGNAL IS MEASURED BY COUNTING A 500 MHZ CLOCK SIGNAL. THE $\pm$ 1 BIT ERROR OF A SINGLE MEASUREMENT IS

$$\Delta R = \frac{c}{2} \cdot \frac{1}{f_{cl}} = 30\ cm$$

MULTIPLE PULSE AVERAGING

THE DIGITIZING ERROR AS WELL AS OTHER STATISTICAL ERROR SOURCES ARE REDUCED BY MULTIPLE MEASUREMENT AVERAGING

$$\varepsilon_n = \varepsilon_1 / \sqrt{n}$$



ε_1 SINGLE MEASUREMENT ERROR

ε_n ERROR OF AVERAGE VALUE

CALIBRATION AND SELFTEST

OFFSET ADJUST AND SYSTEM SELFTEST CAN BE PERFORMED BY TIME GATING THE RECEIVER: EITHER THE TARGET SIGNAL OR THE FIBER COUPLED TEST SIGNAL IS EVALUATED, DEPENDING ON THE POSITION OF THE ELECTRONIC RANGE GATE.

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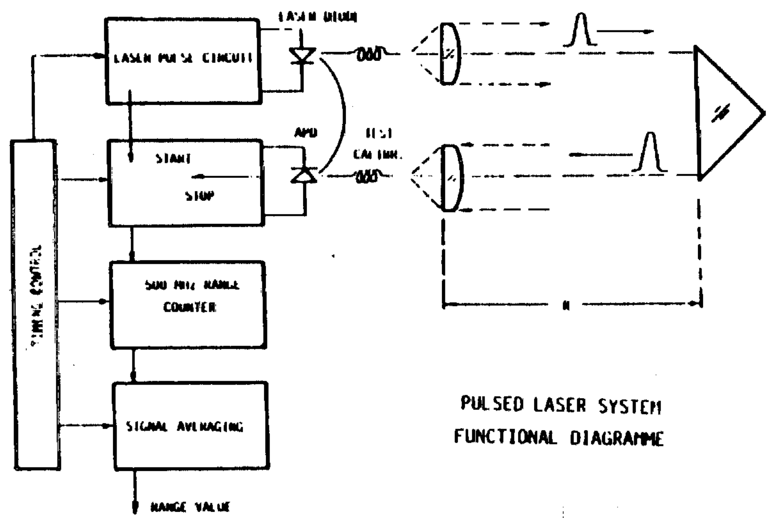
RENDEZVOUS & PROXIMITY SENSORS

PULSED LASER DIODE RANGEFINDER

PRINCIPLE OF OPERATION



PULSED LASER SYSTEM
FUNCTIONAL DIAGRAMME

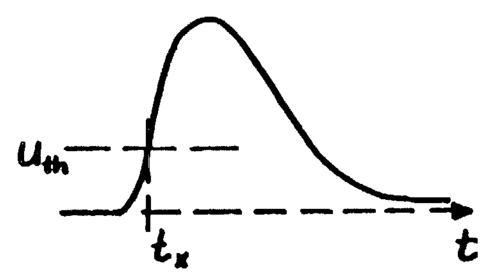
THE RANGE TO THE TARGET R IS DETERMINED BY MEASURING THE TIME-OF-FLIGHT OF A SHORT LASER PULSE ΔT :

RANGE: $R = \frac{c}{2} \Delta T$

TRIGGER TIME EVALUATION: FIXED THRESHOLD DETECTION

THE RISE TIME OF THE LASER PULSE IS VERY FAST (TYP. 1 NS), THEREFORE THE LEADING EDGE OF THE RECEIVER SIGNAL IS USED FOR PRECISE TRIGGER TIME EVALUATION.

RANGE RESOLUTION: $\Delta R = \frac{c}{2} \frac{t_r}{\sqrt{S/N \cdot n}}$



t_r SIGNAL RISETIME
S/N SIGNAL NOISE RATIO
n NUMBER OF MEASUREMENTS

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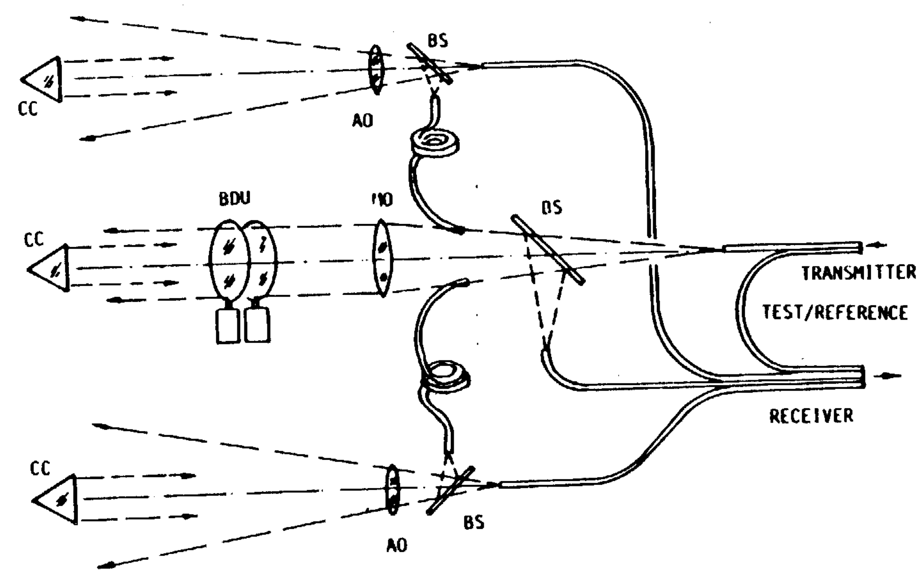
PULSED LASER DIODE RANGEFINDER

CHASER ATTITUDE CONTROL

SHORT RANGE ATTITUDE CONTROL

THE RELATIVE ATTITUDE BETWEEN CHASER AND TARGET SATELLITE CAN BE DETERMINED BY IMPLEMENTION OF FIBER COUPLED AUXILIARY TRANSCEIVERS. THEY MAY BE POSITIONED IN A SIMILAR CONFIGURATION AS SOME AUXILIARY RETROREFLECTORS. THE ORIGINAL LASER SIGNAL IS DELAYED BY FIBERS FOR A TIME INTERVAL LONG ENOUGH TO AVOID INTERFERENCE WITH THE ORIGINAL RANGE SIGNAL. IT IS EASY THEN TO SWITCH FROM RANGE MEASUREMENT TO ATTITUDE CONTROL BY AN ELECTRONIC TIME GATE. IF THE FIBER DELAYS OF THE AUXILIARY TRANSCEIVERS ARE NOT UNIFORM THE ATTITUDE CONTROL CAN BE ACHIEVED BY DETERMINING THE RELATIVE SPACING OF THE RETURN SIGNALS.



Laser Rangefinder with Fibre Coupled Auxiliary Optical Transceiver Systems

BS BEAM SPLITTER
AO AUXILIARY OPTICS
MO MAIN OPTICS
CC CUBE CORNER REFLECTOR
BDU BEAM DEFL. UNIT

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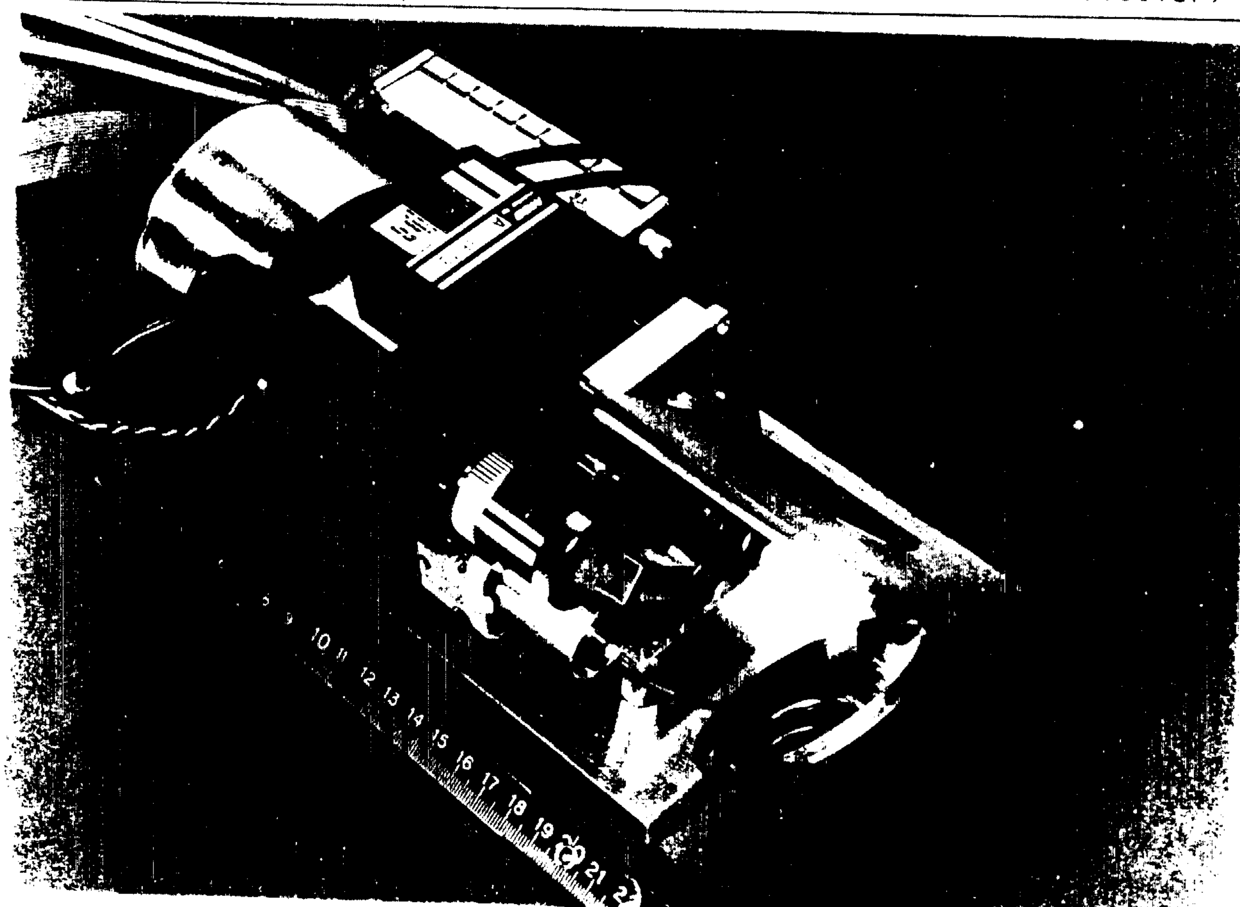
RENDEZVOUS & PROXIMITY SENSORS

Previous MBB Laser Diode Radar Functional Model (ESTEC Contr.)
(Combined cw/pulsed sensor, common scanner+receiver)



PURPOSE: - Verify automated long rate to near-field target tracking ability
- Pulsed system for large FOV acquisition, tracking + coarse ranging
- CW system for precision ranging, range rate + positioning

MBB RENDEZVOUS & PROXIMITY SENSORS

EXAMPLE: MBB PULSED LASER DIODE RADAR SENSOR HEAD, FUNCT. MODEL
(developed on ESTEC Contract)

RENDEZVOUS & PROXIMITY SENSORS

2-AXIS BEAM DEFLECTION (1)

SCANNERS PRINCIPLES/CHARACTERISTICS

Type	Operat. Freq. (Hz)	Feas. Mirror Size (mmxmm)	Max. Angle (°)	ADVANTAGES	DISADVANTAGES
(1) Galvanometer Scanner -Moving Iron Core -Electromagnetic	$0 - 10^3$	$\approx 120^2$	± 60	-Precise Positioning -Great Frequ. Range -All Wave Patterns -Variable fixed position	-Needs locking device to survive launch stress
(2) Torsion Rod/Flex Pivot Scanner (excited by var. magn. field)	100-20000	$\approx 60^2$	± 15	-Long life capability -Simple electromechan. design	-Operates at resonant frequency only ($\pm$ few Hz) -Limited FOV -Sine wave pattern only -No low-frequency operation
(3) Taut Band (Spring) Scanner + permanent magnet	$5 - 10^3$	$\approx 120^2$	± 30	as (2)	as (2) plus -Risk of collapsing at angular excess

RENDEZVOUS & PROXIMITY SENSORS

2-AXIS BEAM DEFLECTION (2)

Type	Operat. Freq. (Hz)	Feas. Mirror Size (mmxmm)	Max. Angle (°)	ADVANTAGES	DISADVANTAGES
(4) Rotating Polygon Mirror Scanner	0-20000	$\leq 100^2$ per segment	$\pm$ 180	-Largest scan angles	-Highest power requirement -Limited positioning precision -Coupling with second axis mirror difficult
(5) Electro-optical (Piezo) + Acousto-optical (Bragg) Beam Deflectors (Voltage drives cell)	0-40000	$\leq 50^2$	≤ 2	-No mechanically moving part -Highest frequency -Low power operation	-Very small FOV only -2-axis scan nearly impossible -limited to very small transm. optics (cw lasers only)

RENDEZVOUS & PROXIMITY SENSORS

CAMERA CONCEPTS INVESTIGATED

(a) VIDEO CAMERA:

- Video data of operations only, requires man

(b) NAVIGATION/TRACKING CAMERA:

- Matrix CCD camera using few light targets (3-4 in L-shape geometry) as reference, range and attitude data derived from size/distance between targets and their relative attitude compared to an "electronic cross-hair"
- In proximity, 3-dimensional reference body (e.g. black cylinder) plus crossed LED-arrays for high-precision measurements

(c) CONTOUR TRACKING CAMERA:

- Matrix CCD camera plus optical processor for contour tracking of significant structures (e.g. solar panels), performance limited, software/processing effort excessive

(d) NAVIGATION CAMERA PLUS LASER FOR ILLUMINATION:

- Similar to (b), using cw or pref. pulsed laser diode for illumination of passive targets

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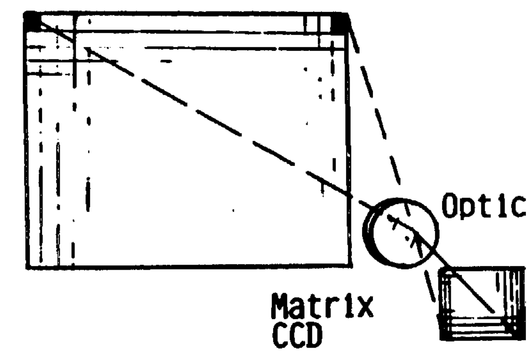
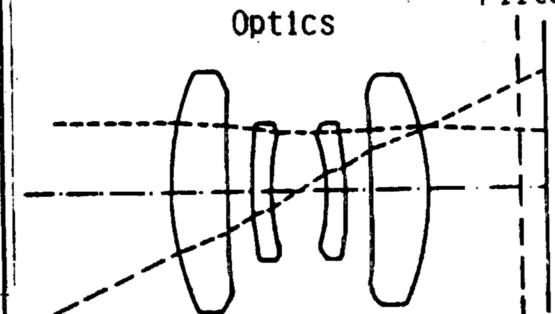
RENDEZVOUS & PROXIMITY SENSORS

MATRIX CCD CAMERA COMBINED WITH LED TARGETS
(Derivative of a Star Tracker Camera)

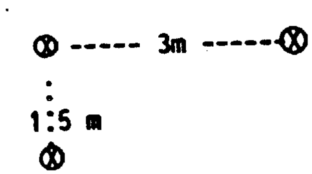
MEASUREMENT PRINCIPLE:

- The search FOV (focal length dependent) is electronically scanned (horizontally and vertically) by the matrix CCD focal plane, relayed onto it with the small telescope in from of it; for background suppression reasons (Earth + Sun on S/C, a narrow-band interference filter may be placed between optics and FP;
- Except for long distances, where for an acquisition the whole TS may be taken as target, the camera processor will discriminate distinct targets on the TS, preferrably arranged in the same L-type configurations as the laser reflectors; for performance and background suppression reasons we strongly recommend to have <u>active</u> light emitting <u>targets</u> (LEDs) which are frequency-coded (orientation recognition, background)
- The camera processor will derive all measurement parameters from the relative distance between the targets resp. their coordinates in the fixed FP pattern
- In the near-field, especially for the relative attitude determination, a 3-dimens. reference body and coded LED arrays will provide very high resolution for all measurement parameters.





Long Distance Target Configur.

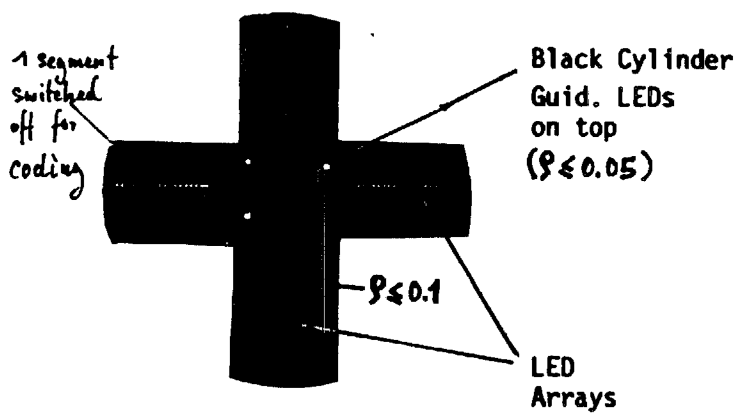


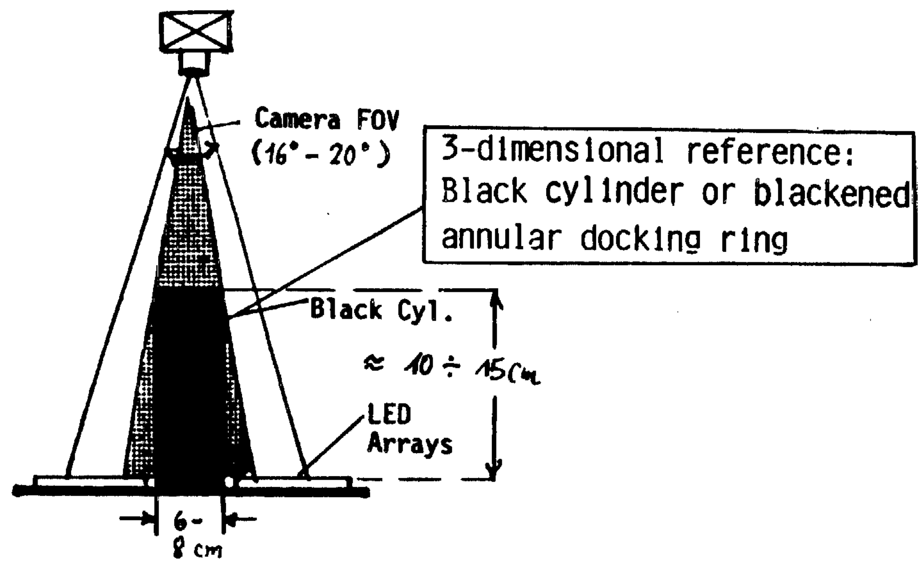
RENDEZVOUS & PROXIMITY SENSORS

MBB FAVOURED CONCEPT OF IMAGE SENSOR TARGETS

NEAR-FIELD MEASUREMENT CONCEPT

Near field (10 m to about 0.3 m):





MATRA ESPACE

RENDEZVOUS & PROXIMITY SENSORS

MATRIX CCD CAMERA COMBINED WITH LED TARGETS

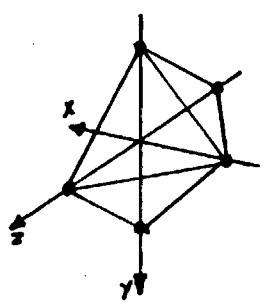
IMAGING / CAMERA SENSOR PRINCIPLE (SRS)

BASICS : IMAGE ANALYSIS OF A DETERMINED PATTERN ON THE TARGET

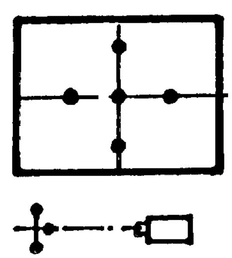
DRIVING CRITERIA :
- MINIMIZATION OF IMAGE PROCESSING ALGORITHMS
- 6 DOF MEASUREMENT (SMALL ANGLES, APPROACH WITHIN A CORRIDOR)

STUDY AREAS :
- OVERALL CONFIGURATION
- PATTERN GEOMETRY
- PROCESSING ALGORITHM
- SENSOR DIMENSIONING (FOCUSING STRATEGY, PHOTOMETRY, OPTICS, MECHANISM, ELECTRONICS).

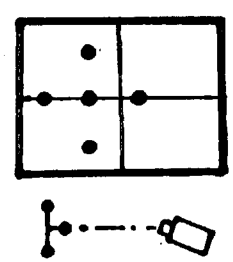
6 DOF MEASUREMENT PRINCIPLE : PYRAMIDAL 5 POINTS PATTERN



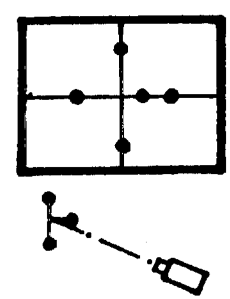
PATTERN SHAPE



NO ERROR

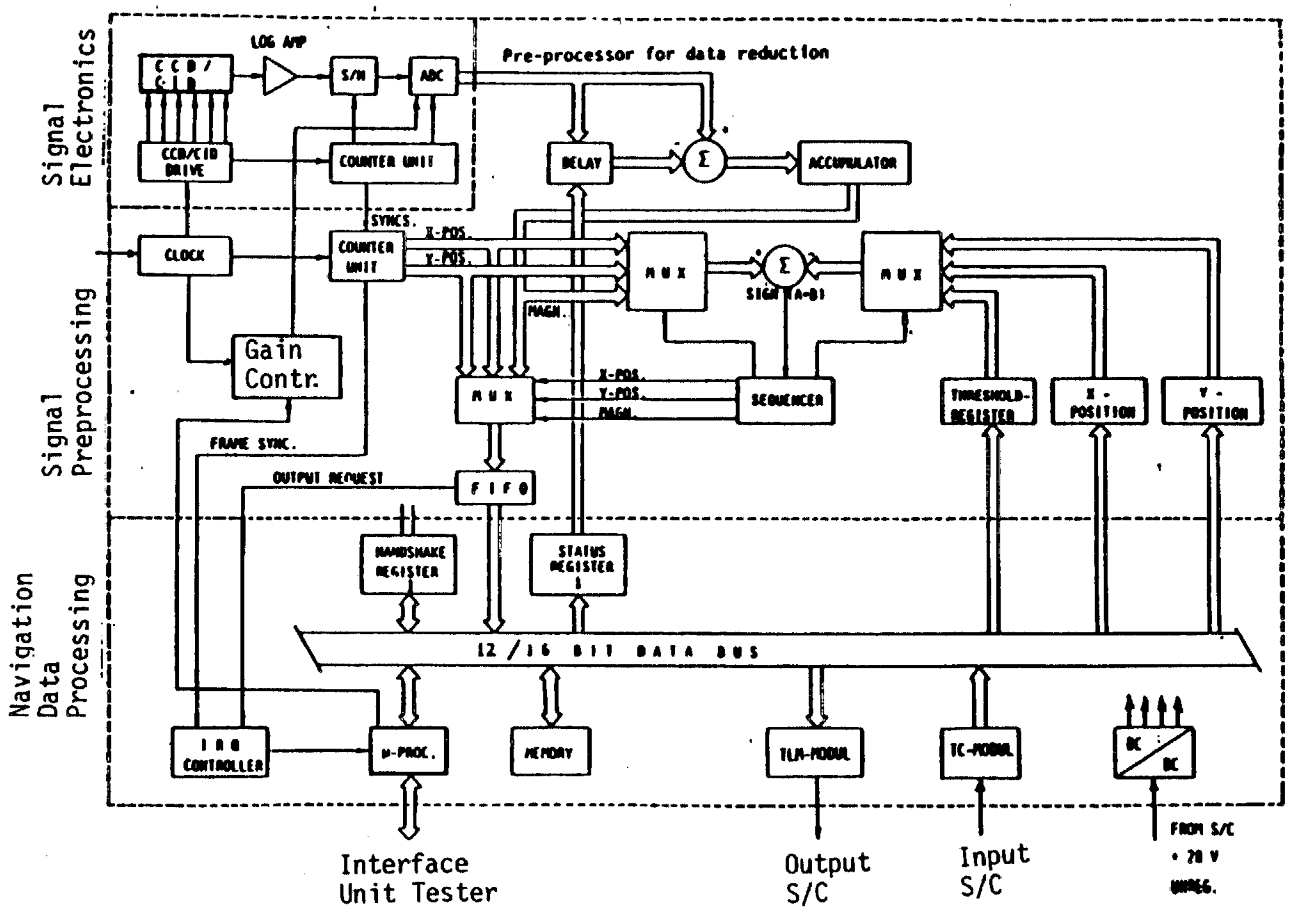


ANGULAR ERROR



ANGULAR AND POSITION ERROR

STDC NAVIGATION CAMERA ELECTRONICS
BLOCK DIAGRAMME



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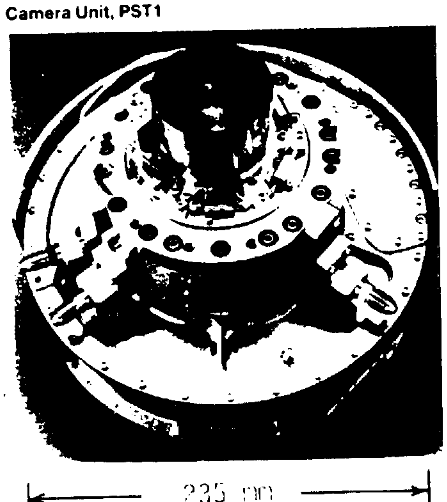
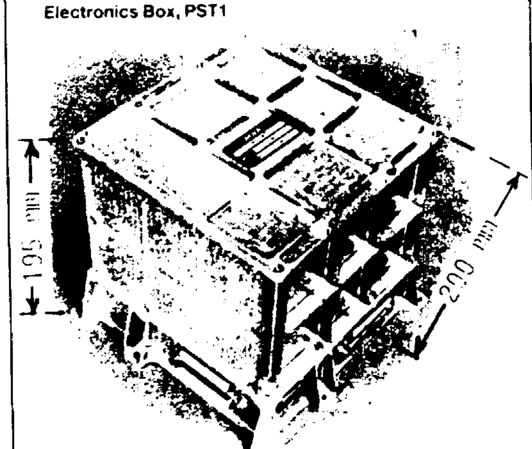
NAVIGATION CAMERA DERIVED FROM PRECISION STAR TRACKER

Camera Unit, PST1



Electronics Box, PST1



Precision Star Tracker PST1.

Field of view:	5.9 × 4.4
Sensitivity:	m_v = −6.5 to 0
Accuracy:	
Pointing:	≤ 2arc sec
Scanning:	≤ 10arc sec
Noise equivalent:	
Pointing:	≤ 1arc sec
Scanning:	≤ 5arc sec at scan rates up to 5 arc min/sec
Magnitude accuracy:	+/−0.25 m_v
Update period:	1 sec
Star acquisition time:	≤ 4 sec
Tracking capacity:	4 targets
Output data:	magnitude, position, HK-data
Reliability (1.5 years mission):	> 0.912
Power consumption:	15 W
Weight: Sensor unit	8 kg
Electronics unit	5 kg

Developed for ROSAT under BMFT contract.

RENDEZVOUS & PROXIMITY SENSORS

GENERAL PERFORMANCE CHARACTERISTICS
OF OPTICAL RVD SENSORS

	PARAMETER	LASER DIODE SENSORS	CAMERA+ACT. TARGETS	COMB.LASER+CAMERA
RANGING	Range R Capability (km)	$0 \div 500km$	$0 \div \approx 12km$ (focal length depend.)	0 - 2km (laser depend.)
	Range Resolution (% of R	≈ 0.1	$\leq 0.1 \leq \Delta R \leq 100m$	$\leq 0.1 \leq \Delta R \leq 100m$
	Range Rate $\dot{R}$ (m/s) - 10km - 100m 10m	 ≈ 10 ≈ 0.01 ≈ 0.001	 (≥ 10) ≈ 0.01 ≈ 0.001	 -- ± 0.05 ≤ 0.01
ANGULAR PERFORMANCE	Bearing Angles-Resol. (°) 100m 10m	 < 0.1 "	 $\pm 1 - \pm 3$ 10^{-3}	 $\leq \pm 0.1$ $\leq \pm 0.1$
	Tilt (pitch)/yaw Angle Resol. (°) 10m 1m	 ± 1	 ± 0.02 ≤ 0.005	 t.b.d. ± 0.5
	Instr. total FOV (°)	± 30 in 2 axis/ ± 25 or 360° in azim./ 50 in elev.	7 x 9 to 10.5 x 14 20 x 25 pref.	11 x 14
	Frame Repetition Time (s)	20 / 30° FOV 1 / 2.5° FOV	0.1	0.1 - 0.2

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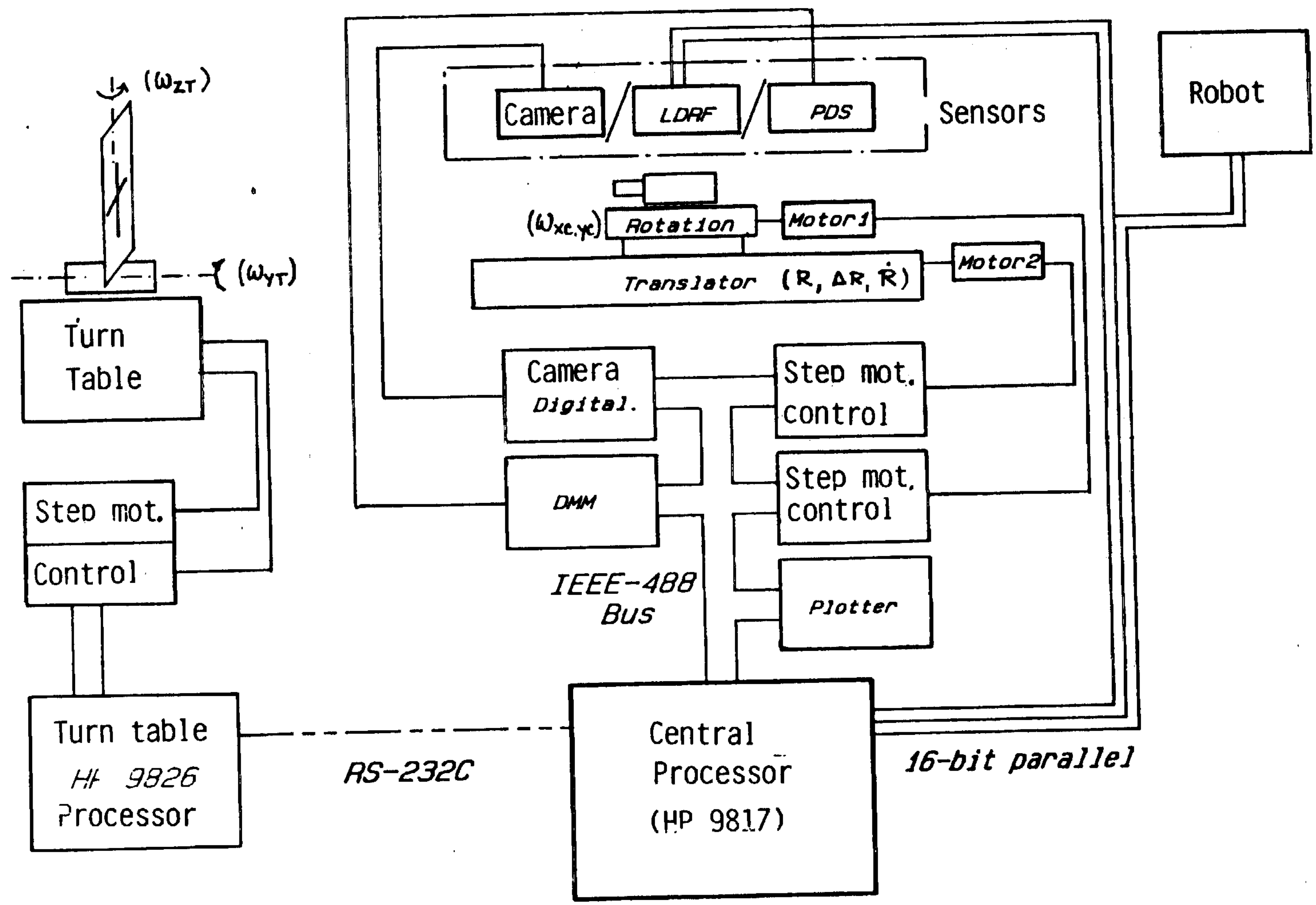
RVD SENSOR TEST FACILITY (at MBB/Ottobrunn)



RENDEZVOUS & PROXIMITY SENSORS

POTENTIAL ROLE OF OPTICAL SENSORS FOR TASKS
BEYOND RVD/PROXIMITY OPERATIONS
for COLUMBUS Type Missions

TASK \ SENSOR	CAMERA SENSORS			LASER DIODE RVD SENSORS	COMBINED LASER (Illum.) + CAMERA
	Video Camera	Navig. Camera	Cont.Tr.C.		
• Video Control Data of Operations	X	(X) FOV limitat.	X	--	((X)) Laser point./FOV limitations
• Data Link (opt) betw. S/C	--	--	--	X (plus mod./dem.)	--
• Relative position keeping monitoring betw. detached S/C	--	(X) (Light targets + camera point. needed)	((X))	X	(X)
• In-orbit Servicing, Telemanipulation, Proximity Operations Support/Monitoring	(X)	(X)	(X)	((X))	((X)) as above

X: fully applicable; (X) applic. with constraints; ((X)) Applic. doubtful

MBB ERNO

RENDEZVOUS & PROXIMITY SENSORS

OPTICAL RVD SENSORS - INTERFACE PROBLEMS

LASER SENSORS + COMBINATION LASER/CAMERA

INTERFACES WITH SENSOR CARRYING S/C	INTERFACE CRITICALITY	
	operational	physical
- Free FOV towards target S/C (varying LOS?)	uncritical?	in case of 360° FOV in azimuth: rotation mount on top plus free FOV in elevation (t.b.d. deg)
- Ressources/Accomm. I/F	--	moderate (dimens., weight, power)
- Approach Trajectory, Orbits	may be rather incisive (in case or large angle deviation from nominal LOS)	defines free FOV accommodation area an carrier S/C ("chaser")
- AOCS Processor I/F	criticality depend on repetition rate (critical at 1s in near field)	harmonisation of format and I/F circuits needed with laser processor
- Data Handling, Telemetry	uncritical	uncritical (1-5 kbps), harmonisation of I/F needed
I/F WITH TARGET S/C		
- Retro Targets on Target S/C	none (passive targets) as long as retros turned towards approaching S/C	Accommodation area needed, size range dependent (targets -3+3- are small, $\leq$ 5cm Ø)

RENDEZVOUS & PROXIMITY SENSORS

OPTICAL RVD SENSORS - INTERFACE PROBLEMS

CAMERA SENSORS

INTERFACES WITH SENSOR CARRYING S/C	INTERFACE CRITICALITY: Operational	INTERFACE CRITICALITY: Physical
- Free FOV towards target S/C (varying LOS)	uncritical	expected to be uncritical
- Resource/Accomm. I/F	N.A.	" " " " because of low-power, compact size
- Approach Trajectory, Orbits	very critical (varying LOS)	pointing mechanism/larger FOV
- AOCS Processor I/F	- timing (repetition) uncritical	Harmonisation of input I/F needed (format, data rates etc.)
- Data Handling, Telemetry	- Timing of intermediate video data - Availability of high data rate link (≥ 50 kbps)	
I/F with TARGET S/C		
- Active/passive Targets	- Switch on/off command required	- Accomm. of 2 target systems on Target Satellite (incl. 3-dimens. reference moderate electronics + power)
- Video Data Transmission (if requ.)	- Interlacing with RVD data - Data Handling	- Data link

RENDEZVOUS & PROXIMITY SENSORS

CONCLUSIONS

- Optical Sensors (CCD Cameras and Laser Diode Sensors) can provide very accurate navigation data until physical docking
- Laser Sensors provide full range capability at simplest processing of data of all RVD sensors
- Camera sensors are superior in the near field and can also provide video data
- The technology status is rather advanced, to nearly functional model status which require modifications
- Further technology development steps are needed prior to space testing in the area of processing/software development and testing and space qualification of essential subunits which require soon iniatiation.
- MW sensors appear to be less promising for addressed ranges, superiority for long ranges (> 20km); because of proven technology, they may serve as primary sensor for ≤ 1km distance and backup for shorter distance.

Effects of Tethers on Rendezvous and Proximity Operations

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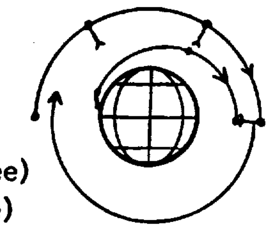
February 1985

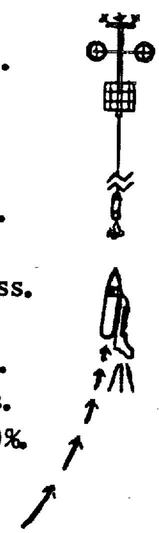
Remote capture (by boom or tether) creates:

Unique Benefits	&	Unique Problems
Fast approach from afar		Hazardous if inaccurate
RCS plume directed away		Hovering in place is costly
Safety (remote capture)		Loads on end masses (~.01 gee)
Large propellant savings		Orbit changes (compensable)



A look at tethered capture of the STS by an SSPE:

- Launch windows for tethered & free-fall docking are similar.
- GPS (in relative position mode) ensures precise approach.
- STS can hover using OMS/RCS (<10 min.) until captured.
- If capture fails, STS can boost to a free-fall docking.
- Capture probe on ET raises safety & payload volume & mass.
- Rigidization can be safely delayed until dynamics damped.
- ET scavenging possible during retrieval, w/low STS scar mass.
- ET GO2/GH2 can be used for contamination-control purging.
- Later STS-deboost reboosts SSPE above SS orbit for phasing.
- Loaded OMVs at tether tip can deboost ET/orb if tether fails.
- Bottom line: a 50 km tether may raise STS throughput by >50%.
  (= Payload + cryos + OMS savings from capture & deboost.)



Some hard questions:

Are scheduled disturbances & H2 contamination acceptable on a major SSPE?
Is a ~400x500 km SSPE orbit acceptable on occasion, when T_{ex} is not high?
Should a tether node be the main SSPE (since STS, OMVs, & OTVs would use it)?
Should NASA seriously consider having such a transport node in the SS Program?
What early tests could best determine whether such a node is feasible?

For more info on related tether topics, see the following (by the author):

Tether-Mediated Rendezvous (3/84, for Martin Marietta, open dist.)
A Scenario for Evolution of Tether Uses on a Space Station (AIAA-84-110-CP, CSI)
Tether Applications in Space Transportation (10/84, IAF 84-438, IAF Cong.)
Some Tether-Related Research Topics (11/84, for JPL & CSI)
Guidebook for Analysis of Tether Applications (to be available from MSFC)
Roles for Tethers on an Evolving Space Station (final report due 4/85, CSI)

27 March 1985

SESSION 7 - DISPLAYS AND HUMAN FACTORS

7-1. "The Geometrical and Symbolic Content of a Perspective Display for Commercial Aviation: Implications for Spatial Proximity Operations Displays" - Stephen Ellis and Michael McGreevey/NASA ARC

7-2. "Use of Perspective Displays for Situational Awareness in Space Station Proximity Operations" - Michael McGreevey and Stephen Ellis/NASA ARC

7-3. Advanced R&T Base Control/Display Technology with Potential for Rendezvous and Proximity Operations" - J. Hatfield and R. Parrish/NASA LaRC; R. Montoya/Research Triangle Institute

7-4. "System Engineering Approach to the Advanced Work Station for Space Station" - John Hussey/Grumman

7-5. Application of Voice Interactive Systems for the Future" - Carolyn Moore and Douglas Moore/VERAC Inc. and John Ruth/McDonnell Douglas Electronics Company

7-6. "THURIS - The Human Role in Space" - S. Hall/NASA MSFC

7-7. "Stereo Video and Display Systems" - Nicholas Shields/ Essex Corporation

The Geometrical and Symbolic Content of a Perspective Display for Commercial Aviation: Implications for Spatial Proximity Operations Displays

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Abstract

Proximity operations of co-orbiting spacecraft require careful monitoring of spatial separation. Although the parameters of interest are different for aircraft and spacecraft operation, the spatial judgments required for proximity operations are similar to those of commercial aircraft pilots flying near other aircraft. We have developed and tested a perspective display for commercial aviation which illustrates how the perspective projection and specific spatial position information may be selected to provide appropriate spatial separation cues. The approach used to create this format provides an example of how the metrical symbology of a perspective display may be designed to facilitate spatial judgments in a specific environment. The design, for example, reflects the usual difference in the range of vertical and horizontal aircraft velocities by a differential scaling of the corresponding axes of the display. Analogous considerations concerning proximity operations should be made to select appropriate format and symbology.

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The Geometrical and Symbolic Content of a Perspective Display for Commercial Aviation

1. Introduction

The optimal presentation of spatial separation information to operators of aircraft or spacecraft is a major consideration in cockpit design. This optimization is important if the designer wishes to allow the operators maximum flexibility of use while requiring minimum training time. Furthermore, this optimization is particularly important in situations where the cost of operation is high, such as on-orbit use of the Shuttle Remote Manipulator System (RMS) which might cost as much as $50,000/hour.

Accordingly, displays of spatial information must be designed so that the required perceptual tasks match the users spatial abilities. We have attempted to establish this match through the use of a 2D display with perspective to present airline pilots with their 3D traffic situation. Our definition of a perspective situation display for this application provides an example of carefully designed spatial display and concretely illustrates two classes of generic issues associated with the design of such displays: 1) those raised by the projective character of their geometry and 2) those raised by the symbolic content of displayed objects and text.

2. Perspective Situation Display for Commercial Aircraft

2.1. Purpose of the Display

Perspective displays were defined primarily to provide a commercial airline pilot with a summary of his air traffic situation. These displays were intended to

allow the pilot to monitor his surrounding air traffic to cross-check the safe operation of the air traffic control system. However, the display also served as a general situation awareness indicator since it included navigation information in addition to air traffic. In the experiments we have conducted the pilot's general task was to detect violations of standard spacing and select avoidance maneuvers necessary to achieve it, if either vertical or horizontal separation was violated. The detailed procedures and results of these experiments may be found in several published reports.(Palmer, *et. al.*, 1981; Smith, Ellis, and Lee, 1982; Ellis, and McGreevy, 1983; Ellis, McGreevy, and Hitchcock, 1984; McGreevy and Ellis, 1984) The following outline will focus primarily on the perspective display format that was used.

2.2. Symbolic Content

The symbolic content of the display is generally the data defining the graphics objects that are projected on the display screen.

☛ Realistic wire frame models of all aircraft were used as aircraft symbols such that the current position of each craft was under the nose of the symbol.

☛ A grid which provided a general sense of 3-D space was located relative to the pilot's ownship current altitude and aligned with his aircraft's velocity vector to provide a consistent relative horizontal separation metric.

☛ Future position of all aircraft in 1 minutes was shown by predictor lines protruding from the the aircraft symbols nose.

☛ Reference lines were dropped onto the grid from the nose of each aircraft symbol and from its future position to indicate horizontal separation and to unobtrusively eliminate the ambiguity of aspect otherwise inherent in perspective projections.

➨ Ownship's current altitude was indicated by 'x's on the reference lines and 1000 ft relative altitude units shown by tics on these lines. ownship relative altitude 'x's and 1000 ft tics

2.3. Projective Geometry

The projective aspects of the display primarily refers to the effects of the matrix transformations on the objects that are displayed.

➨ The perspective projection was defined to allow changes in major parameters of the projection in such a way that these changes would not cause objects in a defined space to pass out of the field of view.

➨ All aircraft size's were scaled relative to ownship so as to keep ownship a constant size regardless of the parameters of the perspective projection. This relative scaling insured the visibility of all aircraft in all projection conditions.

➨ The vertical scale of the display was adjusted so that regardless of perspective parameters used, the ownship symbol would remain a distance in screen coordinates above the reference grid that was proportional to altitude above ground level.

➨ The adjustment in the vertical axis of the displayed space was made to reflect the convention that 1000 feet of vertical separation provided the same spacing as 3 naut. mi. horizontal separation. The resulting scaling insured that vertical separation of less than 1000 feet will be perceptually distinct from separations greater than 1000 feet.

➨ The direction of the principle viewing vector from the geometric eye point was rotated with respect to the reference grid to provide improved spatial sense.

3. General Considerations for Perspective Displays

Experiments comparing perspective with matched plan-view displays have shown that perspective displays can have a major impact on practical decisions such as how to avoid conflicting air traffic. Results have shown, for example, that pilots may make greater use of the vertical dimension for avoidance maneuvering and make fewer blunders or unnecessary maneuvers (Ellis, McGreevy, and Hitchcock, 1984). It is, thus, clear that perspective displays can have practical utility for the presentation of spatial information.

The specific design of a perspective display is highly dependent upon the task for which it is to be used. However, some elements of the display symbology are generic to all perspective displays.

For example, one generic aspect of the symbology adopted for the aircraft situation display is that special aspects of the symbology, such as the altitude tics and the grid orientation were defined "relative to the pilot's ownship". This principle is used to allow convenient relative spacing judgments and to minimize problems such as control reversals due to perceptual confusion of display orientation. Another general aspect of the design of the perspective display was the careful use of symbolic aids, such as the reference lines on the ends of the predictors, to remove ambiguities of attitude intrinsic to geometric projections.

Significantly, both of these design features are not dependent upon the specific hardware technique used to implement the displays but are intended to clarify the informational content of the symbology that is presented. Considerations like these will be important for the design of spatial display formats for the Shuttle and Space Station.

4. References

Ellis, Stephen R., McGreevy, Michael W. Influence of a perspective cockpit traffic display format on pilot avoidance maneuvers. *Proceedings of the Human Factors Society, 27th Annual Meeting*, October 10-14, 1983, Norfolk Va., pp 762-766.

Ellis, Stephen R., McGreevy, Michael W., and Hitchcock, R. Influence of a perspective cockpit display format on pilot avoidance maneuvers. Proceedings of the AGARD Aerospace Medical Panel Symp. on Human Factors Considerations in High Performance Aircraft. Williamsburg, VA, April 30 - May 4, 1984.

Palmer, Everett, Jago, S.J., Baty, D.L., O'Conner, S.L. Perception of horizontal aircraft separation on a cockpit display of traffic information. *Human Factors*, 1980, *22*, 605-520.

McGreevy, Michael W., Ellis, Stephen R. Direction judgement errors in perspective displays. Proceedings of the 20th Annual Conference on Manual Control. June 12-14, 1984.

Smith, J.D., Ellis, Stephen R., Lee, Edward. Perceived threat and avoidance maneuvers in response to cockpit traffic displays. *Human Factors*, 1984, *26(1)*, 33-48.

THE SYMBOLIC AND GEOMETRIC CONTENT OF A PERSPECTIVE SPATIAL DISPLAY

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I. Introduction: the cockpit traffic display example

A. Natural presentation of spatial information
B. Classes of variables in perspective displays
1. Symbology
2. Geometry
C. The intended use for cockpit traffic displays

II. Definition of a perspective cockpit traffic display

A. Symbolic content
B. Projective geometry
C. Developmental sequence of illustrations

III. Implications for rendezvous & proximity operations displays

A. Pivotal role of intended use
B. General requirements for perspective displays
C. Window on the "synthetic universe"

February 20, 1985

SYMBOLIC CONTENT

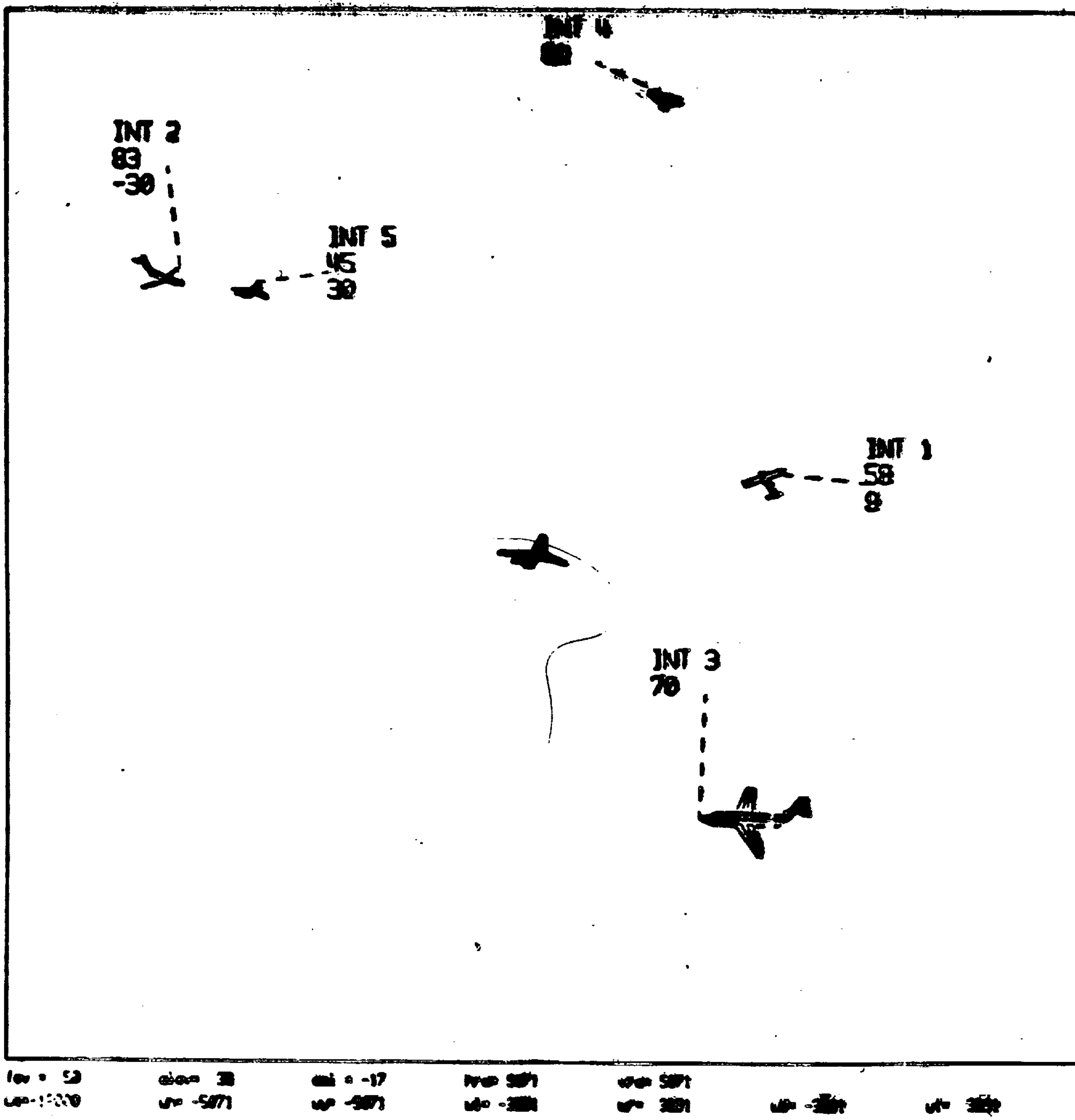
- realistic wire frame models
- grid for 3D spatial sense
- reference lines clarifying position & orientation
- ownship relative metrical aids

PROJECTIVE GEOMETRY

- perspective projection keeping objects in view
- separate scaling of object size
- differential vertical/horizontal scaling
- skewed viewing for improved spatial sense

FIG(4)

Aircraft seen in perspective

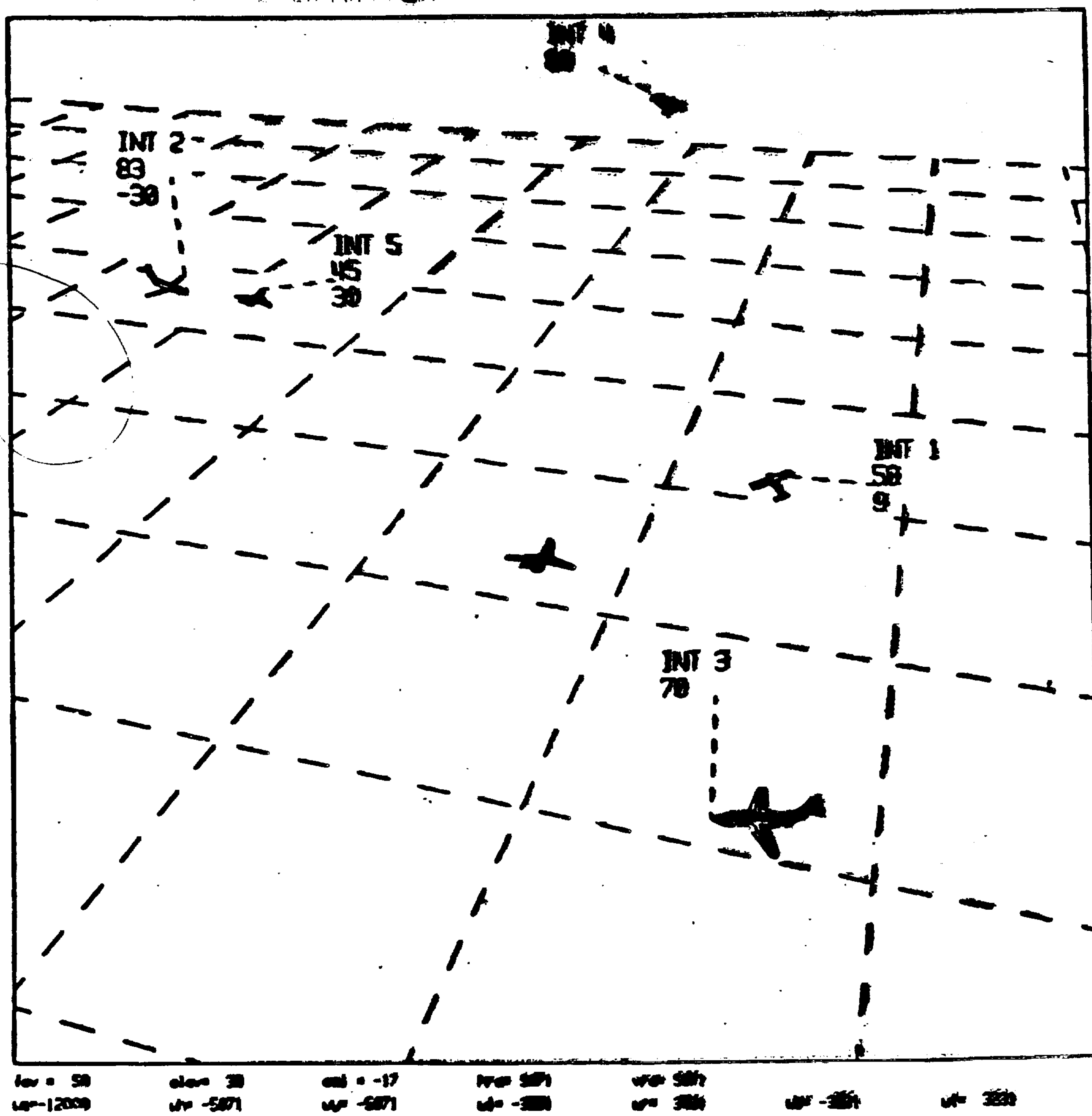


NO. ____

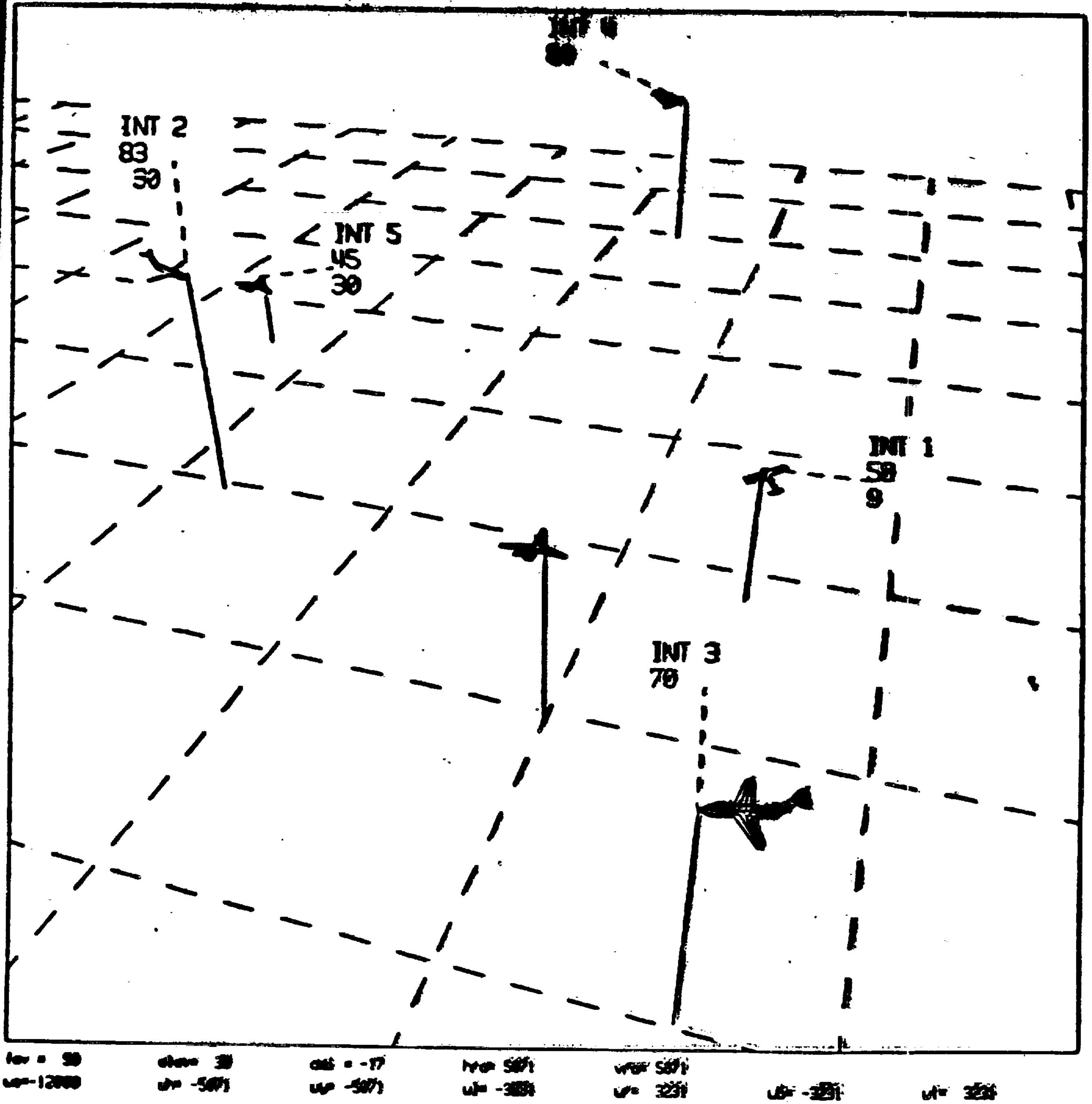
NAME ____

FIG(5)

Scene with horizontal reference grid added.



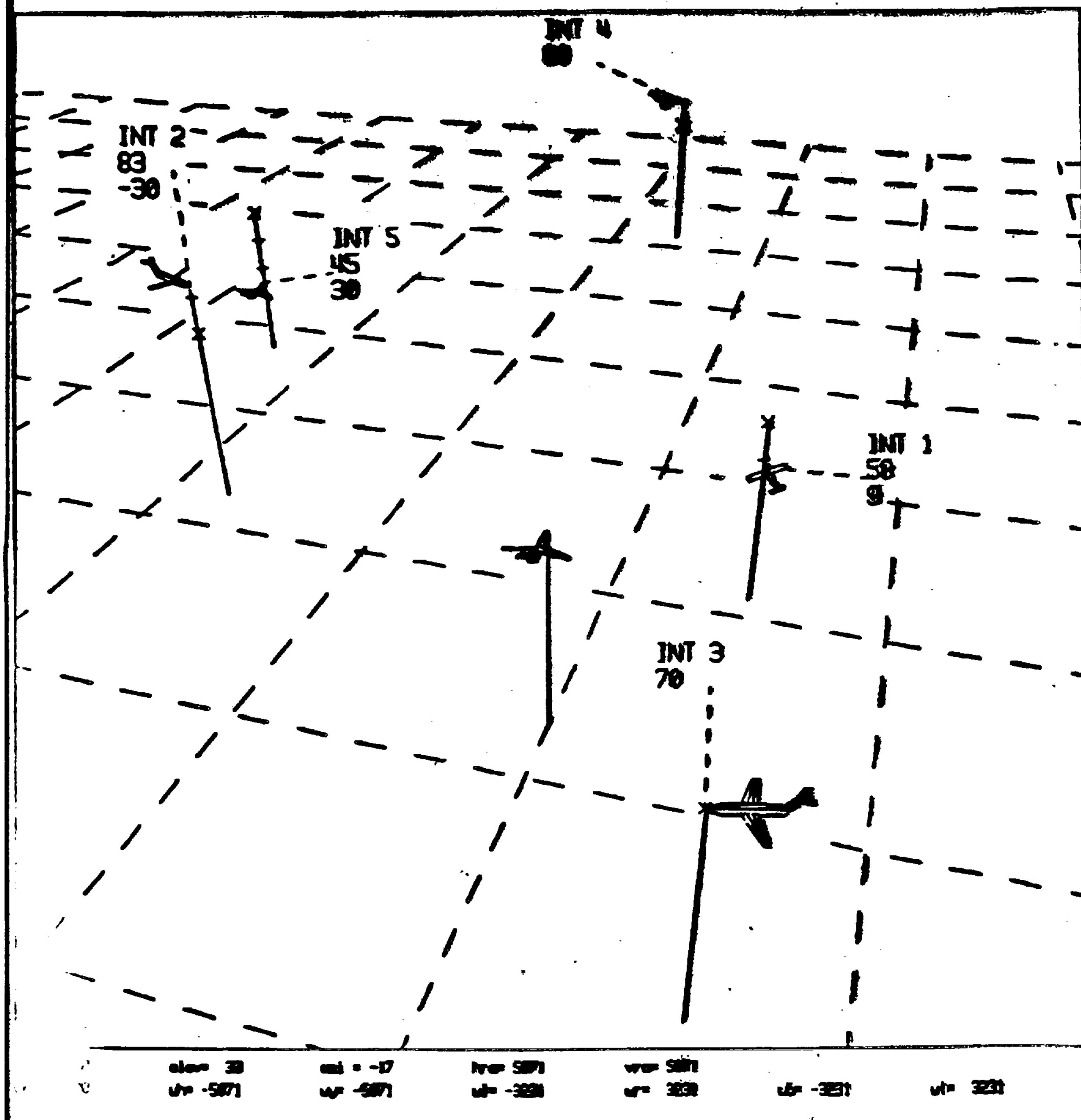
FIG(6)
Scene with noseline showing grid position.



NAME

NO.

FIG(7)
Scene with metrics added to noselines.



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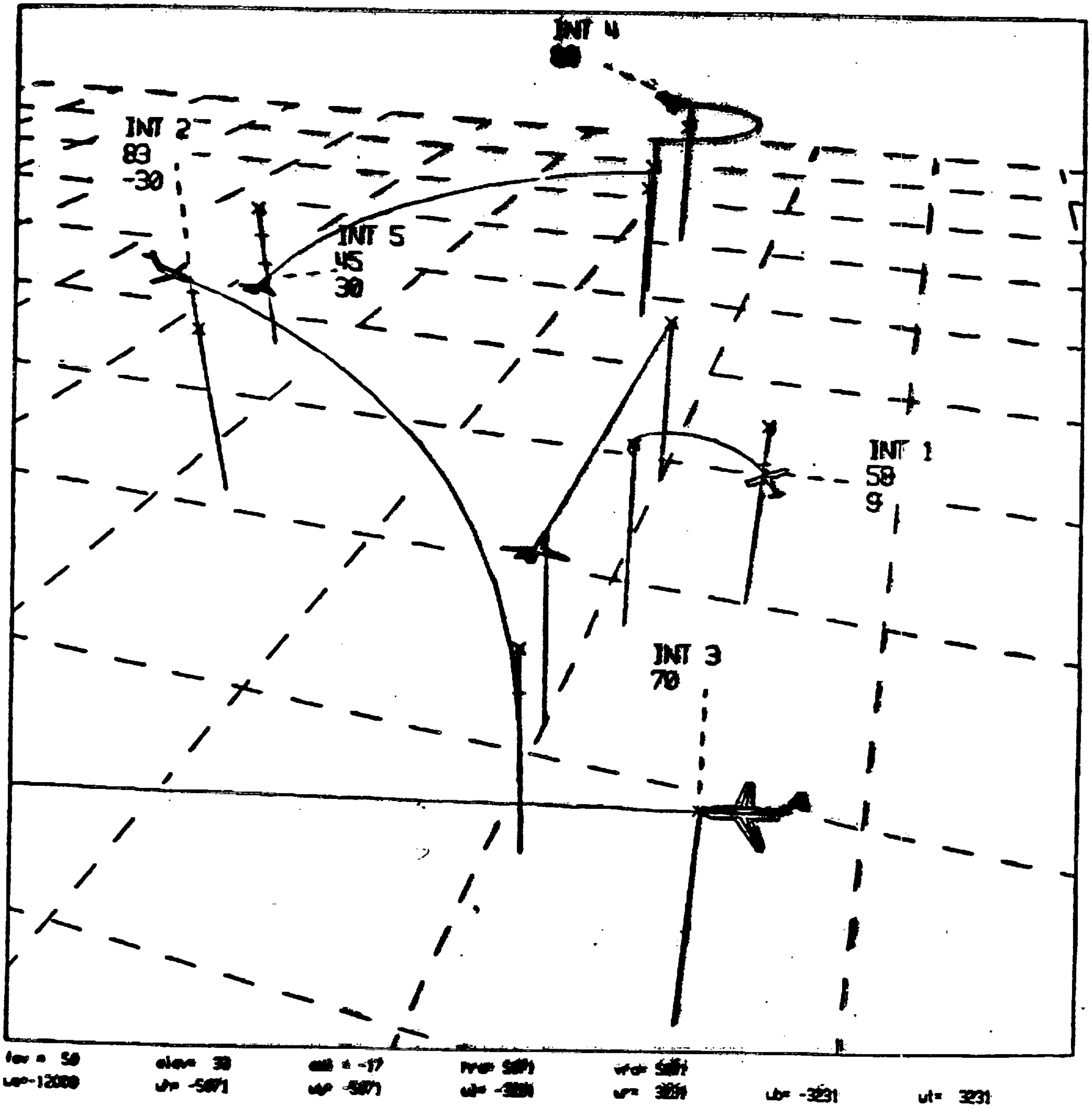
NAME

FIG(8)
Scene with predictors added



7-14 C-5

Use of Perspective Displays for Situational Awareness in Space Station Proximity Operations

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ABSTRACT

Proximity operations around the space station will involve complex and changing elements and activities that will make it difficult for crewmembers to maintain adequate situational awareness. This may result in loss of productivity, and increased risk of data loss, damage, or injury. Spatial displays can provide a visual representation of the relationships among payloads, manipulators, co-orbiters, space station elements, and associated regions of operation, and thus promote situational awareness. This can reduce mission costs by allowing simultaneous, independent activities to proceed without unexpected conflicts.

Spatial displays can be implemented as text, diagrams, maps, perspective views, or virtual environments. Each has its inherent advantages and disadvantages. To see something 'in perspective' is to see it from a particular viewpoint. Perspective is fundamental to direct viewing and to pictorial spatial displays, including standard video, stereo, holographic, and varifocal mirror display techniques. Consequently, perspective design issues are fundamental to spatial display design. It is misleading to call any visual display 'true 3D' because ultimately all visual information is projected to the eyes and is seen in perspective.

Experimental results indicate that spatial judgements in perspective displays are influenced by the relative locations of the design eyepoint and the user's eye position, and differences between the two-dimensional display image and the implied three-dimensional scene. Other experiments have shown that format design can have a significant effect on task performance. Thus, display size, position and perspective geometry must be designed together with task-specific symbology and metric aids to assure optimal information transfer.

Evaluation of the role of situational awareness in proximity operations, and recognition of the potential costs of inadequate awareness, may encourage the use of spatial situation displays. The ideal spatial situation display for proximity operations might use perspective for an integrated view of spatial relationships, supplemented by a text display of exact parameters. This suggests that the initial space station data processing system should include adequate hardware and software for computer graphics.

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Use of Perspective Displays for Situational Awareness in Space Station Proximity Operations

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Proximity operations around the space station will involve complex and changing elements and activities that will make it difficult for crewmembers to maintain adequate situational awareness. This may result in loss of productivity, and increased risk of data loss, damage, or injury. Spatial displays can provide a visual representation of the relationships among payloads, manipulators, co-orbiters, space station elements, and associated regions of operation, and thus promote situational awareness. This can reduce mission costs by allowing simultaneous, independent activities to proceed without unexpected conflicts. Perspective should be used to display spatial relationships and associated text displays should provide precise parameter values.

Perspective displays should be used

1) to show all vehicles and their predicted future positions as a function of maneuvers, orbital mechanics and atmospheric drag;
2) to display protection zones around fragile sensors or other equipment;
3) to indicate active sensors, manipulators, and payloads, and their associated fields of view or areas of activity;
4) to coordinate the various transfer and service vehicles, maneuvering units and remote manipulators to avoid accidental plume impingement, collision, or contact with protruding antennae or solar panels;
5) for monitoring payload installation, maintenance, and the three-dimensional analog of airport ground operations;

Spatial information displays can be implemented as alphanumeric lists of parameters, schematic diagrams, planviews or maps, perspective views, or even synthesized visual environments. The displays that are easiest for designers to implement are usually the most difficult for users to integrate and interpret, probably require excessive training and may induce error. The ideal spatial situation displays should use perspective for ease of interpretability supplemented by text for precision.

1) TEXT
 Typical information presented:
  coordinates, relative angles, separation distances, rates;
 Advantage(s):
  exact parameter values displayed;
  easy for designer;
 Problem(s):
  formats often cluttered and confusingly similar;
  difficult for user to integrate spatial information,
   especially concerning dynamic multiple objects;

2) SCHEMATIC DIAGRAMS (eg. PWI, TCAS)
 Typical information presented:
  selected, low resolution, highly encoded relationships;
 Advantage(s):
  useful for rapid response in one-on-one conflicts with simple solutions;
  consistent with low resolution sensor data;
 Problem(s):
  limited spatial representation capabilities (eg. no objects or trends);
  inadequate for situations that involve dynamic multiple objects;

3) MAPS AND PLAN VIEWS (eg. planview CDTI or in-plane rendezvous maneuver display)
 Typical information presented:
  planar map with out-of-plane dimension (eg. aircraft altitude) encoded;
  object types often encoded;
 Advantage(s):
  clear communication of in-plane position;
 Problem(s):
  difficult for user to integrate out-of-plane information,
   especially concerning dynamic multiple objects;

4) PERSPECTIVE VIEWS (eg. three-dimensional situation display)
 Typical information presented:
  a pictorial analog of the space of interest;
 Advantage(s):
  compatible with natural human spatial abilities,
   including visual understanding of spatial configurations;
  allows use of intuitive -rather than encoded- symbols;
 Problem(s):
  projection ambiguities must be resolved;
  annotation required to display exact parameter values;

5) ENVIRONMENTS
 Typical information presented:
  simulated visual world or "visual telepresence";
 Advantage(s):
  puts the user inside the displayed space;
 Problem(s):
  computationally intensive;
  requires specialized display equipment;

To see something 'in perspective' is to see it from a particular viewpoint. Perspective is fundamental to direct viewing and to pictorial spatial displays, including standard video, stereo, holographic, and varifocal mirror display techniques. Consequently, perspective design issues are fundamental to spatial display design. It is misleading to call any visual display 'true 3D' because ultimately all visual information is projected to the eyes and is seen in perspective.

The common feature of stereo, holographic and varifocal mirror techniques is the presentation of multiple perspectives. Much of the same pictorial effect can be achieved using standard computer graphics displays by tracking the position of the viewer and correspondingly altering the perspective geometry. Whichever of these techniques is used to display spatial information, perspective design issues are fundamental.

1) Stereo consists of two perspective views seen simultaneously, one by each eye. Much of the benefit of stereo can be obtained by monocular motion parallax, that is, multiple perspectives over time.

2) A hologram stores multiple perspectives which are sampled according to eye position. Its main advantage is to allow multiple simultaneous viewers. The unsampled views and the associated computation are wasted. Computer-generated holograms are computationally intensive and currently cannot be generated at a rate adequate for situation displays. A non-holographic computer-generated perspective display can be programmed to display an appropriate perspective for any viewing position, providing a similar effect more economically.

3) The varifocal mirror technique involves sweeping out a virtual image, one two-dimensional slice at a time. This virtual object can be viewed from different positions, providing different perspectives. Its advantages are similar to those of holography but its spatial resolution is currently very low.

The synthetic nature of computer-generated pictorial spatial displays allows augmentation of viewpoint, perspective, scaling, and symbology which can supplement or replace out-the-window or video views.

1) A perspective display is better than a window for overall situation awareness because the synthetic scene may be viewed from an arbitrary point, providing a global view from the most useful direction.

2) By adjusting the perspective geometry and scaling together, the apparent proximity of objects and vastness of space can be made appropriate for the task at hand. Viewing from nearby with a wide angle 'lens' is much like natural vision and makes space seem vast. Viewing from far away with a telephoto 'lens' can present roughly the same region of space but present a view where near and far objects are more nearly the same size.

3) Unequal scaling of symbols or dimensions, and color encoding, can be used for selective emphasis.

4) The overlay of computer generated spatial metrics onto out-the-window scenes or video views would enrich the otherwise cue-impoverished visual environment. The relative depths and separations of objects can be made unambiguous by use of metric aids such as grid planes, separation rulers, and predictors. Metric aids should match the task parameters, as when thousand-foot tic marks are used in an airspace display.

Experimental results indicate that direction judgements in perspective displays are influenced by the relative locations of the design eyepoint and the user's eye position, and differences between the two-dimensional display image and the implied three-dimensional scene. Other experiments have shown that format design can have a significant effect on task performance. Thus, display size, position and perspective geometry must be designed together with task-specific symbology and metric aids to assure optimal information transfer. Surprisingly, it may be undesirable to put the eye at the design eyepoint (also known as the station point).

The location of a pictorial spatial display in the workstation can affect spatial judgement accuracy because the position and size of the screen, and the perspective geometry of the display, all combine to influence the image that is ultimately projected to the user's eyes. If the user were required to be at the station point, the geometry of the displayed space would be severely restricted by the position and size of the display screen. The visual angle subtended by a small, instrument-panel-mounted CRT would a require 'telephoto' projection which could distort the apparent spatial relationships among displayed objects.

Experiments indicate that display users interpret perspective displays as if they were windows, causing a characteristic bias of interpretation. Matching the projections to this expectation by putting the eye at the station point does not eliminate judgement biases, however, because users also are influenced by the two-dimensional projection of three-dimensional spatial relations. There is also experimental evidence of a perspective bias in natural vision which can be cancelled out by a perspective correction. This is equivalent to putting the station point off the user's eye position.

RECOMMENDATIONS

Proximity operations around the space station will involve complex and changing elements and activities that will make it difficult for crewmembers to maintain adequate situational awareness. This may result in loss of productivity, and increased risk of data loss, damage, or injury. Spatial displays can provide a visual representation of the relationships among payloads, manipulators, co-orbiters, space station elements, and associated regions of operation, and thus promote situational awareness. This can reduce mission costs by allowing simultaneous, independent activities to proceed without unexpected conflicts. Perspective should be used to display spatial relationships and associated text displays should provide precise parameter values.

A document should be written, or accumulated from existing documents, which specifies in detail all the spatial parameters of interest in proximity operations.

Included should be:

(a) navigation and control sensor positions, fields of view, resolutions, and update rates
(b) traffic and speed zones
(c) expected paths of vehicles
(d) dimensions of protection zones near sensitive payloads and devices
(e) plume impingement constraints
(f) manipulator reach and clearance
(g) locations to be accessed by MMU
and related information.

Space station computational facilities should be made adequate to meet the hardware and software requirements of effective spatial displays.

1) High resolution, bit-mapped CRTs should be standard equipment because they are useful for spatial displays, can be shared with other display software, are easily reconfigurable, and allow for future workstation upgrades.
2) Computer graphics hardware which can compute homogeneous matrix transformations rapidly enough to support dynamic symbology should be standard equipment.
3) A dedicated general purpose processor should be provided as a display controller.
4) A video signal mixer should be available for combining video camera images and computer-generated metric aids.

The role of situational awareness in proximity operations should be further evaluated. The cost of training and contingency preparation, and the potential for data loss, damage or injury should be weighed against the cost of increasing situational awareness and the potential savings due to increased productivity.

BIBLIOGRAPHY

Billings, C. E. and E. S. Cheaney, eds.,, "Information transfer problems in the aviation system," NASA Technical Paper 1875, National Aeronautics and Space Adminsitration, September 1981.

Carlbom, Ingrid and Joseph Paciorek, "Planar geometric projections and viewing transformations," *Computing Surveys*, vol. 10, no. 4, pp. 465-502, December 1978.

Ellis, Stephen R., Michael W. McGreevy, and Robert J. Hitchcock, "Influence of a perspective cockpit traffic display format on pilot avoidance maneuvers," *Proceedings of the AGARD Aerospace Medical Panel Symposium on Human Factors Considerations in High Performance Aircraft. Williamsburg, Virginia*, April 30-May 4, 1984.

Falzon, Pierre, "Display structures: compatibility with the operators' mental representation and reasoning process," *Proceedings of the Second European Annual Conference on Human Decision Making and Manual Control*, June 2-4, 1982.

Farber, James and Richard R. Rosinski, "Geometric transformations of pictured space," *Perception*, vol. 7, pp. 269-282, 1978.

Foley, J. D. and A. Van Dam, *Fundamentals of Interactive Computer Graphics*, Addison-Wesley, 1982.

Jones, Loren F., H. J. Schrader, and J. N. Marshall, "Pictorial display in aircraft navigation and landing," *Proceedings of the I.R.E.*, pp. 391-400, April 1950.

McGreevy, Michael W., "A perspective display of air traffic for the cockpit," *Proceedings of the 18th Annual Conference on Manual Control*, Dayton, Ohio, June 10, 1982.

McGreevy, Michael W., "A perspective display of air traffic for the cockpit ," MS report, University of California, Berkeley, May 1983.

McGreevy, Michael W. and Stephen R. Ellis, "Direction judgement errors in perspective displays," *Proceedings of the 20th Annual Conference on Manual Control*, Sunnyvale, California, June 12-14, 1984.

McGreevy, Michael W. and Stephen R. Ellis, "Format and basic geometry of a perspective display of air traffic for the cockpit," *NASA Technical Paper*, to be published in 1985.

Newman, W. M. and R. F. Sproull, *Principles of Interactive Computer Graphics*, McGraw-Hill, New York, 1979. 2nd edition

Pepper, R. L., R. E. Cole, and E. H. Spain, "The Influence of Camera Seperation and Head Movement on Perceptual Performance under Direct and TV-Displayed Conditions," *Proceedings from SID*, vol. 24, no. 1, pp. 73-80, 1983.

Roscoe, Stanley N., Scott G. Hasler, and Dora J. Dougherty, "Flight by periscope: Making takeoffs and landings; The influence of image magnification, practice, and various conditions of flight," *Human Factors*, no. 8, pp. 13-40, February 1966.

Smith, J. D., S. R. Ellis, and E. C. Lee, "Perceived threat and avoidance maneuvers in response to cockpit traffic displays," *Human Factors*, vol. 26, no. 1, pp. 33-48, February 1984.

Use of Perspective Displays for Situational Awareness in Space Station Proximity Operations

Michael Wallace McGreevy
Stephen R. Ellis

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Moffett Field, CA 94035

Issues Underlying Display Design

- Human role in automated systems
  - "Computers will do it all."
  - "Everything is done by the numbers."
  - "Humans monitor the computers."
  - "Humans are the source of error."

- Human as processor
  - Garbage in, garbage out
  - Applications notes
  - Insight, judgement, flexibility

- Human-machine interaction
  - Real costs of non-design
  - Digital and analog components

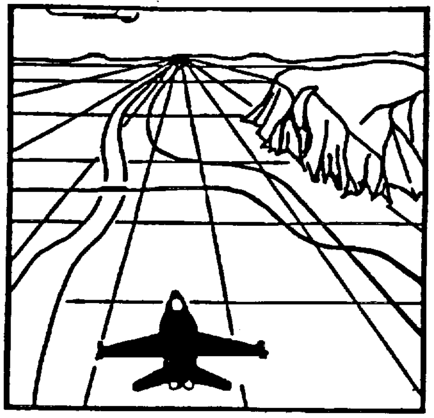
- The need for human factors
  - Exact but unintegrated information
  - The minimum display requirement
  - Artists as designers
  - Realism vs. real information
  - Spatial information transfer

Situational Awareness in Prox Ops

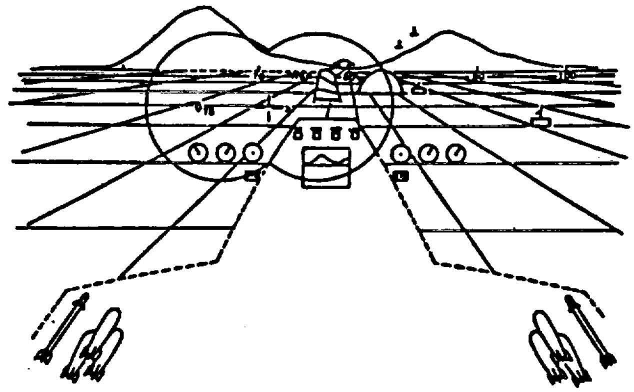
- Why bother?
  - Coordination of parallel operations
  - Crew, structure and vehicle safety
  - Data integrity
  - Efficiency and order
  - Productivity

- Dimensions of the problem
  - Diversity of payloads and vehicles
  - Multiple degrees of freedom
  - Multiple frames of reference
  - Potential for disorientation

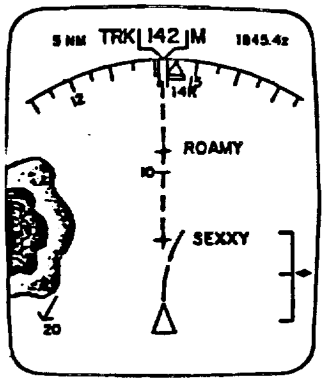
The Integration Dimension



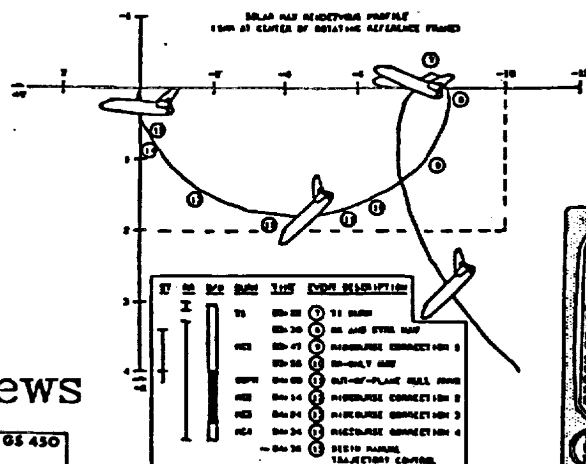
• Perspective views

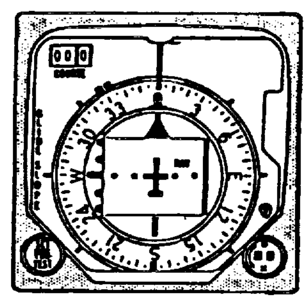


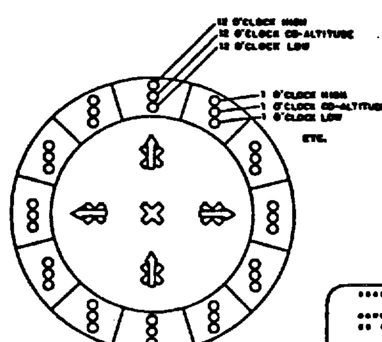
• Environments



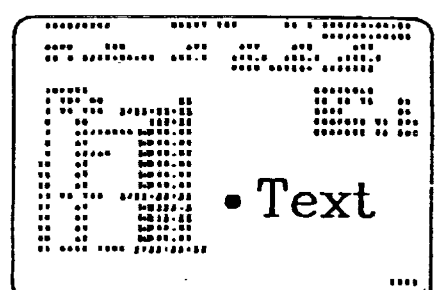
• Maps and planviews



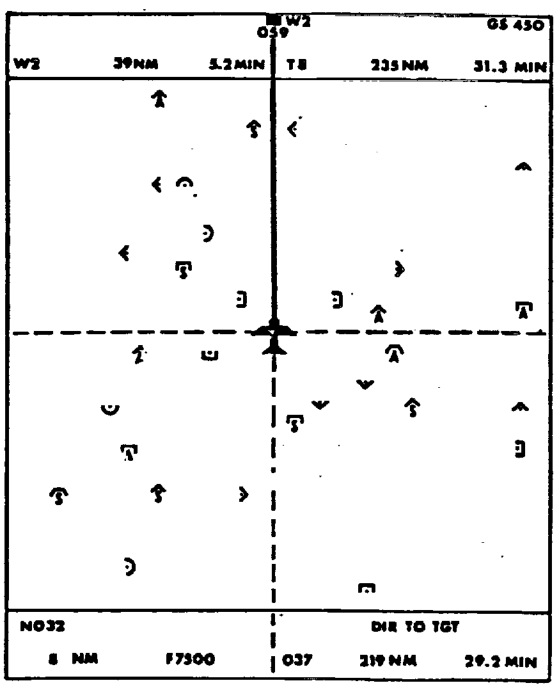




• Schematic diagrams



• Text

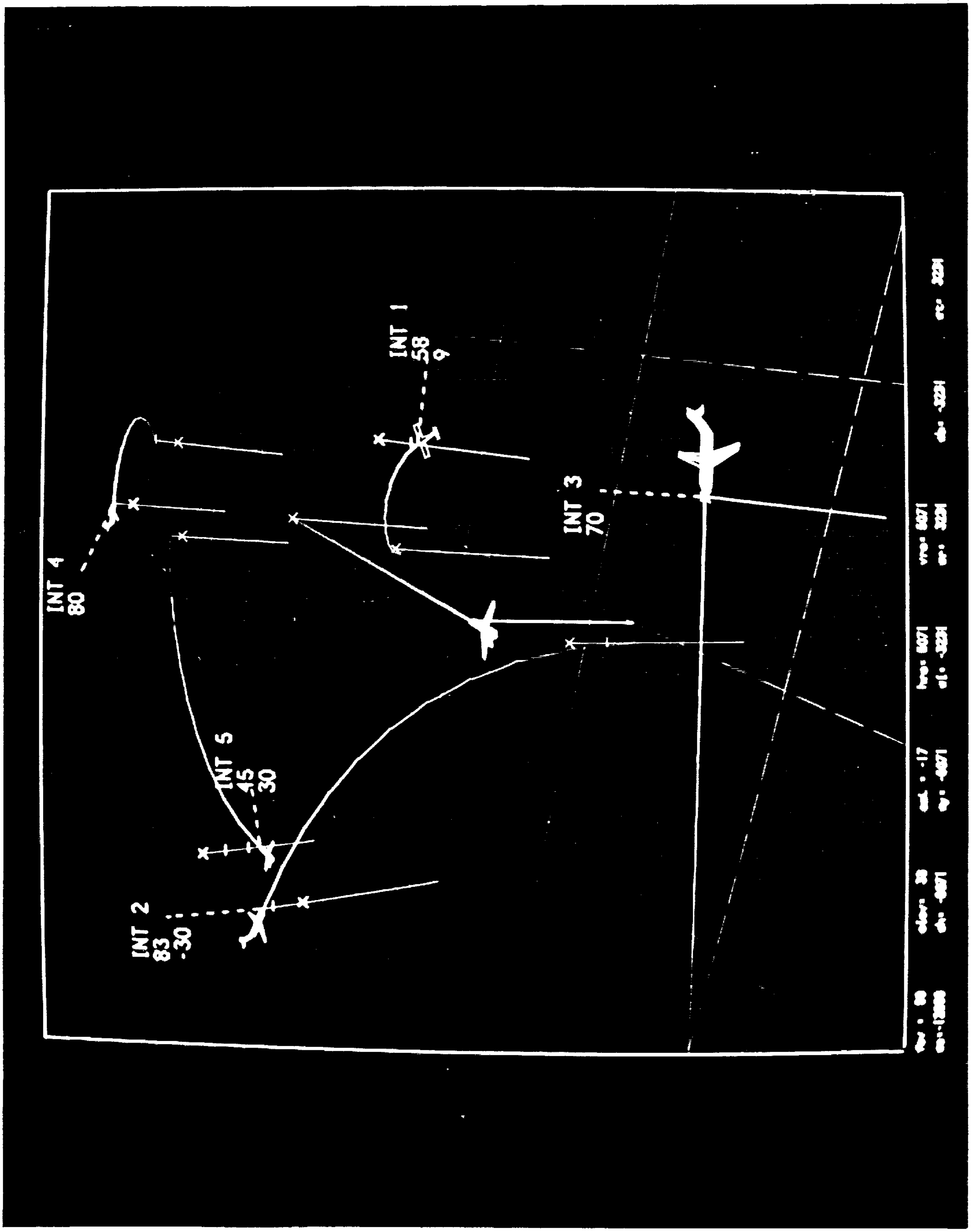


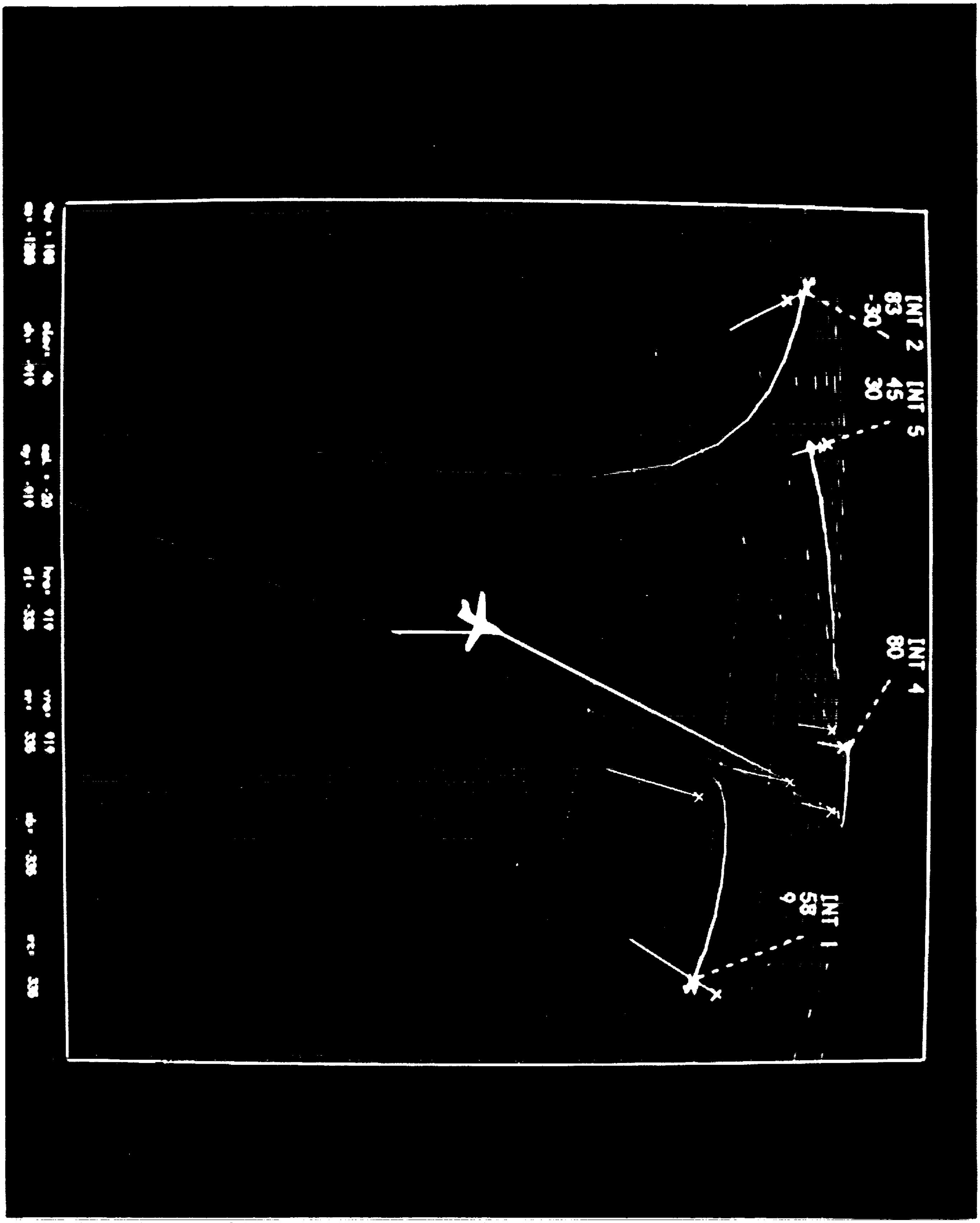
Spatial Information Instruments

- The Integration Dimension
  - Text
  - Schematic diagrams
  - Maps and planviews
  - Perspective views
  - Environments

- The Myth of the Truly 3D Display
  - Vision is projective.
  - Stereo
  - Holographic
  - Volumetric
  - Motion Parallax
  - Perspective is fundamental.

- Use of perspective displays
  - Integrated view of spatial relationships
  - Symbolic and metric aids
  - Selective emphasis and scaling
  - Arbitrary, multiple views


INT 1
58
9
INT 2
83
-30
INT 3
70
INT 4
80
INT 5
45
30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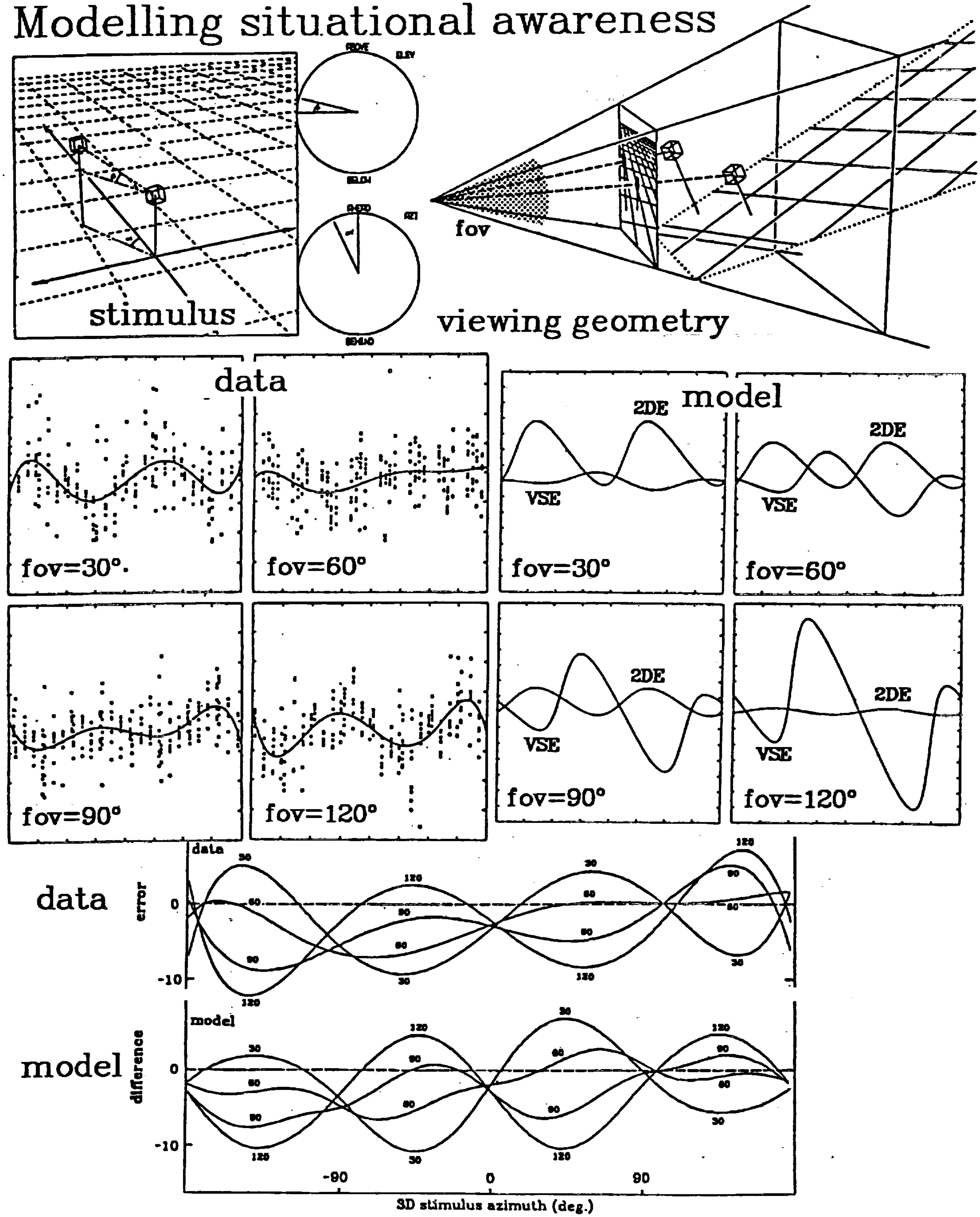
INT 1
58
9

Modelling situational awareness



Spatial Displays for Prox Ops

- Does format matter?
  - Thinking of the world as flat
  - Thinking about space
  - Perspective geometry

- Modelling situational awareness
  - Relative position
  - The influence of projection
  - Not seeing around corners
  - Minimizing sinusoidal error
  - Display placement

- Recommendations
  - Evaluate situational awareness
  - Evaluate the alternative
  - IOC: provide graphics support
  - Document spatial parameters
  - Measure spatial information transfer

ADVANCED R&T BASE CONTROL/DISPLAY TECHNOLOGY WITH POTENTIAL FOR RENDEZVOUS & PROXIMITY OPERATIONS

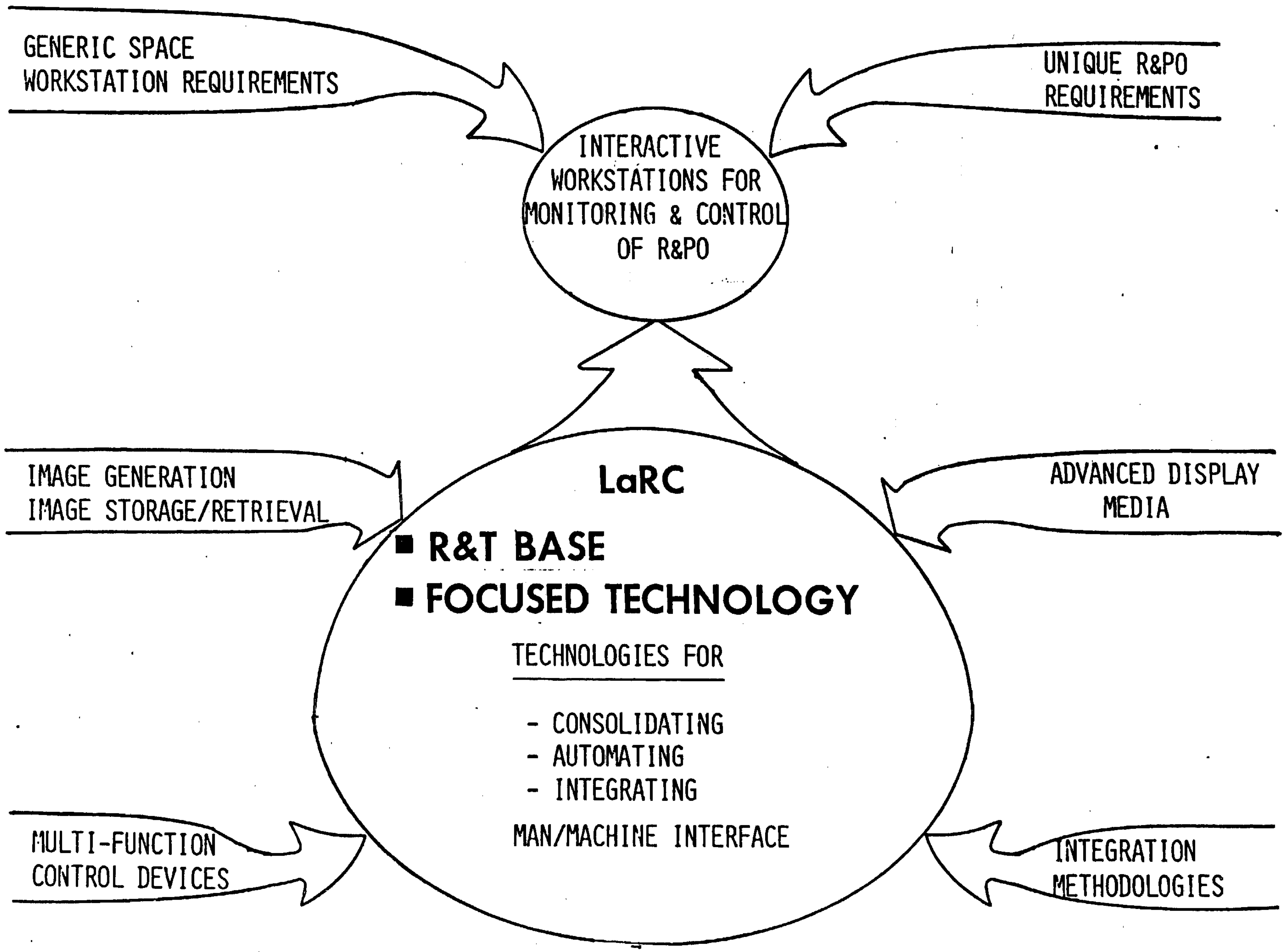
- J. J. HATFIELD, NASA LANGLEY RESEARCH CENTER
  - OVERVIEW/OBJECTIVES
- R. J. MONTOYA, RESEARCH TRIANGLE INSTITUTE
  - RASTERGRAPHIC DISPLAY GENERATION TECHNOLOGY
- R. V. PARRISH, NASA LANGLEY RESEARCH CENTER
  - DISPLAY MEDIA/INFORMATION MANAGEMENT TECHNOLOGIES
  - SUMMARY/CONCLUSIONS
- MOTION PICTURE/VIDEO SEGMENTS (TIME PERMITTING)

OVERVIEW/OBJECTIVES

JACK J. HATFIELD

NASA-LANGLEY RESEARCH CENTER

AREAS OF CONTROL/DISPLAY R&D



R & T BASE EFFORT

CREW STATION TECHNOLOGY RESEARCH

OBJECTIVES:

(1) PROVIDE LONG-TERM, HIGH-PAYOFF OPTIONS FOR THE DESIGNERS OF FUTURE AERONAUTICAL AND AEROSPACE VEHICLES THROUGH RESEARCH ON ADVANCED ELECTRONIC CONTROL & DISPLAY TECHNOLOGY

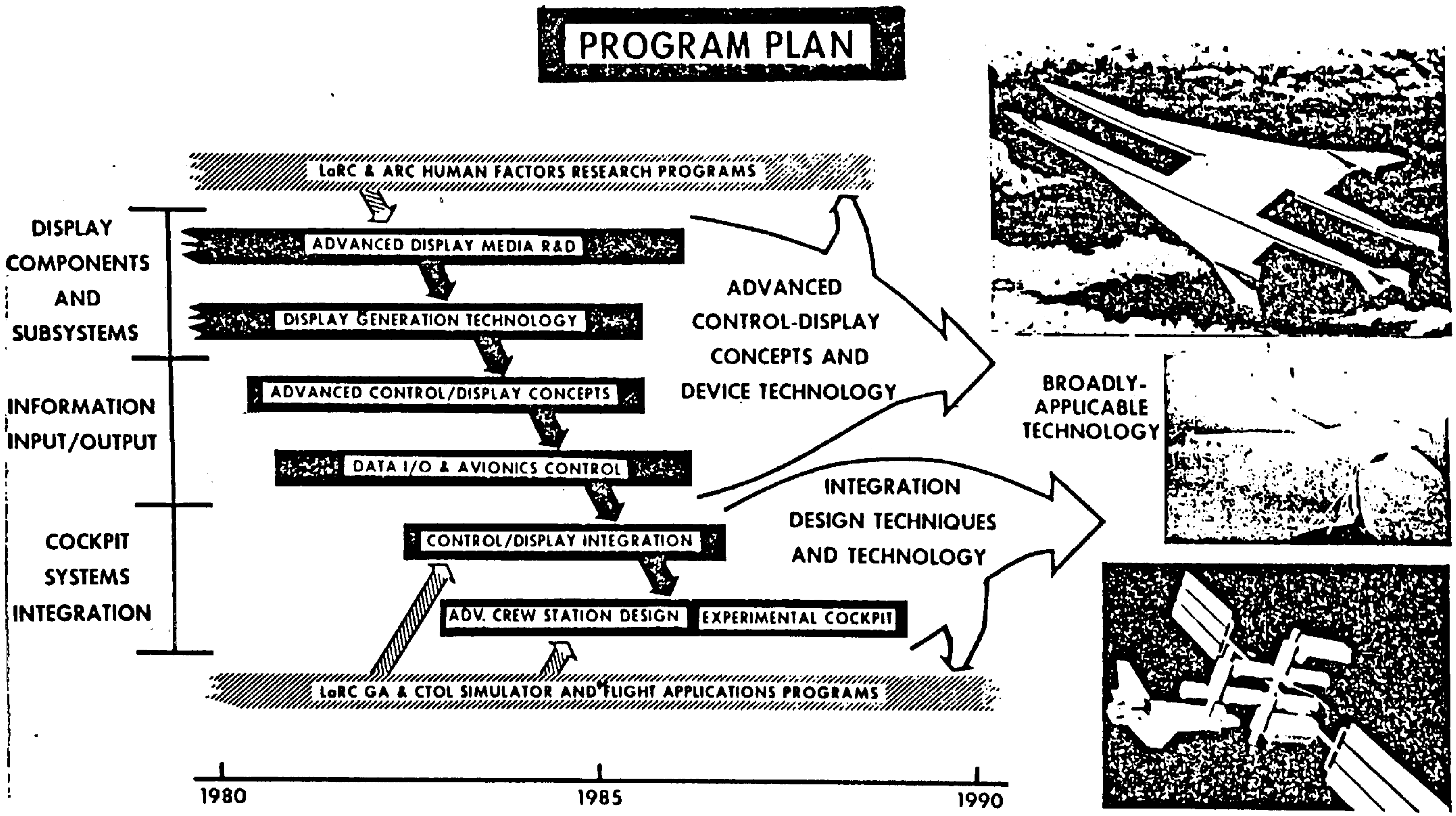
(2) PROVIDE ADVANCED ENABLING TECHNOLOGIES, KNOWLEDGE BASES, AND DESIGN CRITERIA/METHODOLOGIES IN THE FOLLOWING AREAS:

- o DISPLAY COMPONENTS AND SUBSYSTEMS
- o INFORMATION INPUT/OUTPUT
- o COCKPIT SYSTEMS INTEGRATION

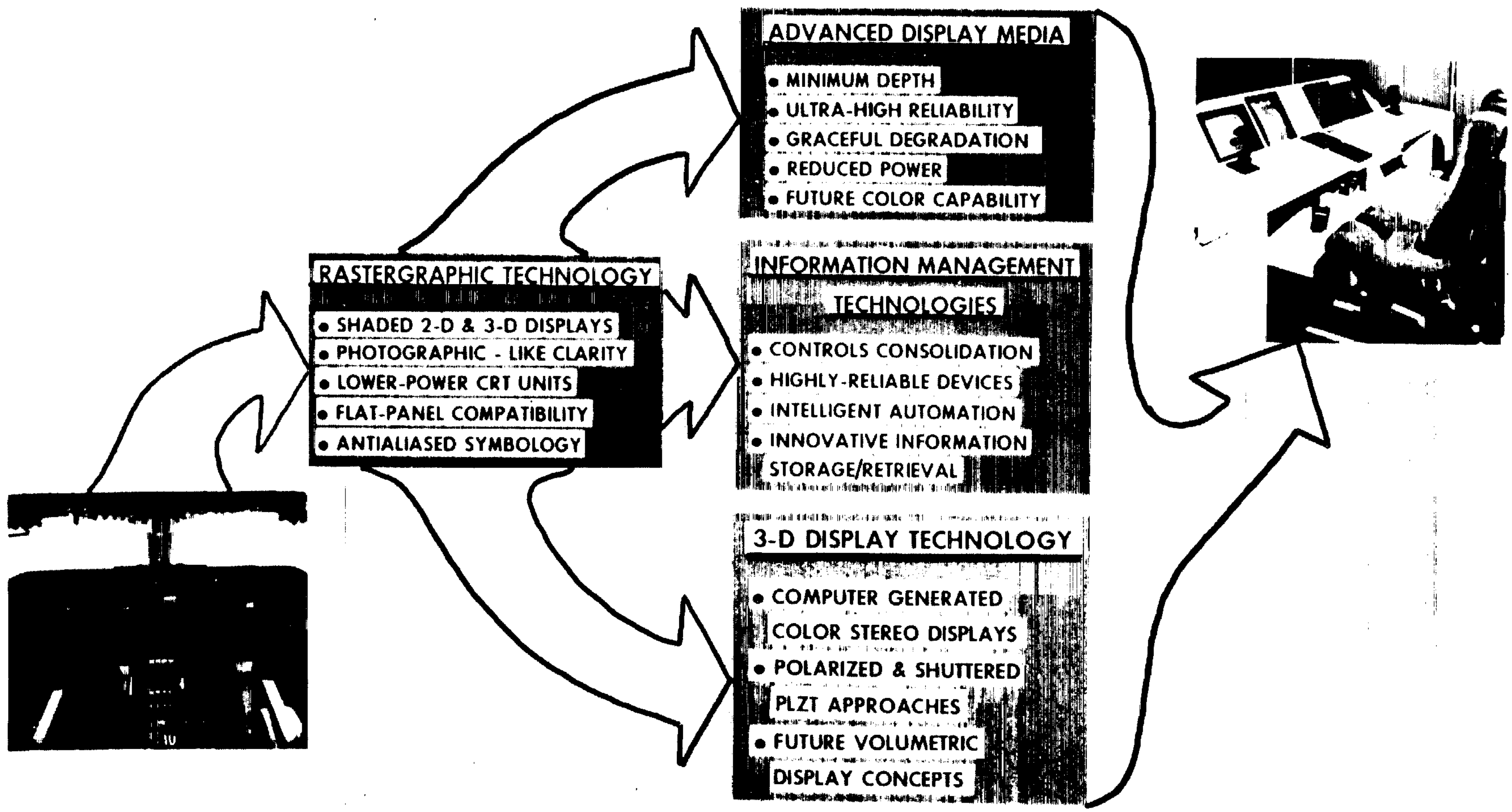
(3) EMPHASIZE LONG-LEAD-TIME, HIGH-RISK TECHNOLOGIES WHICH CAN BEST COMPLEMENT THE ACTIVITIES OF THE AEROSPACE INDUSTRY

R & T BASE EFFORT

CREW STATION TECHNOLOGY ELEMENT



RESEARCH THRUSTS IN CREW STATION TECHNOLOGY

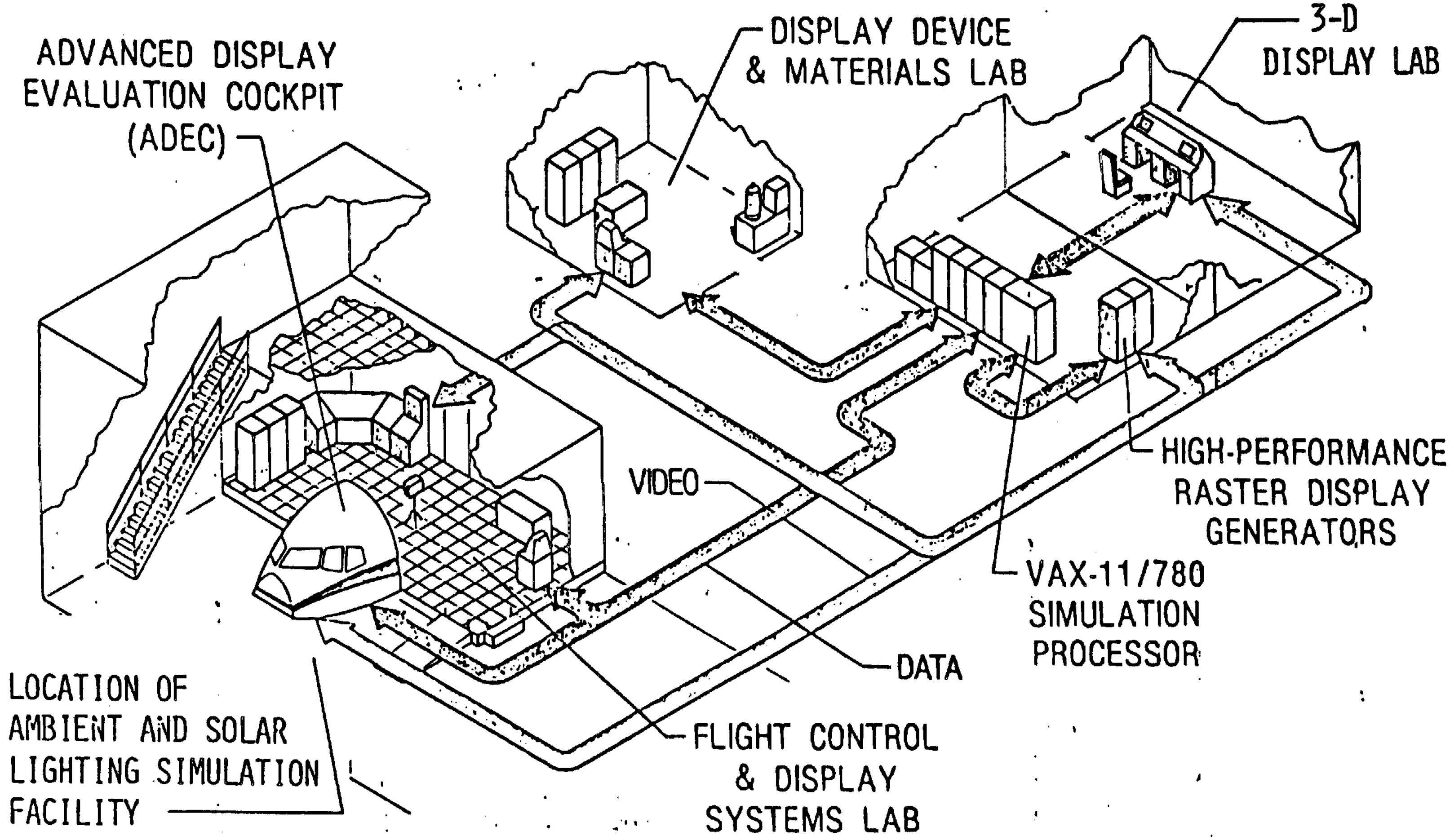


NASA
L-84-10,927

POTENTIAL BENEFITS

- UNCLUTTERED CREW STATION
- REDUCED WORKLOAD
- IMPROVED PILOT/SYSTEM PERFORMANCE
- REDUCED COST-OF-OWNERSHIP
- IMPROVED RELIABILITY/MAINTAINABILITY
- REDUCED TRAINING/CROSSTRAINING
- REDUCED DESIGN PRODUCTION COSTS

CREW STATION SYSTEMS RESEARCH LABORATORY



RASTERGRAPHIC DISPLAY GENERATION TECHNOLOGY

R. JORGE MONTOYA

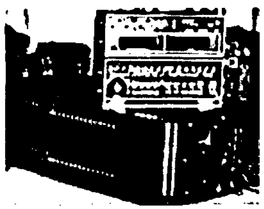
RESEARCH TRIANGLE INSTITUTE

RASTERGRAPHIC DISPLAY GENERATION TECHNOLOGY: PLANS AND OBJECTIVES

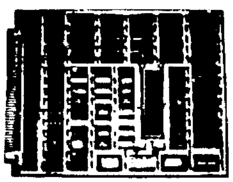
- DEVELOP RASTERGRAPHIC TECHNOLOGY TO PROVIDE GENERATION SPEEDS HERETOFORE AVAILABLE WITH STROKE TECHNOLOGY
- PROVIDE ALL ADVANTAGES INHERENT IN RASTERSCAN TECHNOLOGY
  - EFFICIENT LARGE AREA SHADING
  - WINDOWING CAPABILITIES
  - PRIORITIZATION OF SYMBOLOGY
  - SIMPLER/LOWER COST DEFLECTION SYSTEM
  - LESS DISPLAY POWER CONSUMPTION
  - PHOTOGRAPHIC-LIKE CLARITY
- PROVIDE UNIQUE RESEARCH FLEXIBILITY
  - VARIABLE RESOLUTION & FRAME RATES
  - COMPATIBILITY WITH CRT AND FLAT-PANEL DISPLAYS
- UTILIZE RESEARCH INVESTIGATIONS BY A NASA/UNIVERSITY/INDUSTRY TEAM
- APPLY/EVALUATE THE TECHNOLOGY IN A LAB AND SIMULATOR ENVIRONMENT IN CONJUNCTION WITH ADVANCED MEDIA

CREW STATION ELECTRONICS R & T BASE
SIGNIFICANT ACCOMPLISHMENT

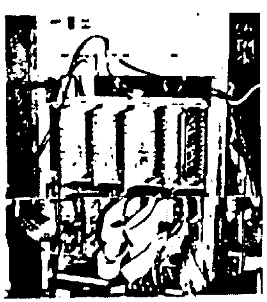
RASTERSCAN COLOR GRAPHIC DISPLAY TECHNOLOGY ADVANCE ENHANCES ADVANCED CONCEPT SIMULATORS AT LARC, ARC, LOCKHEED AND NADC



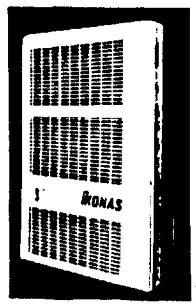
RESEARCH TRIANGLE INSTITUTE- SYSTEMS/ APPLICATIONS SOFTWARE



IKONAS GRAPHICS SYSTEMS- COMMERCIAL VENTURE

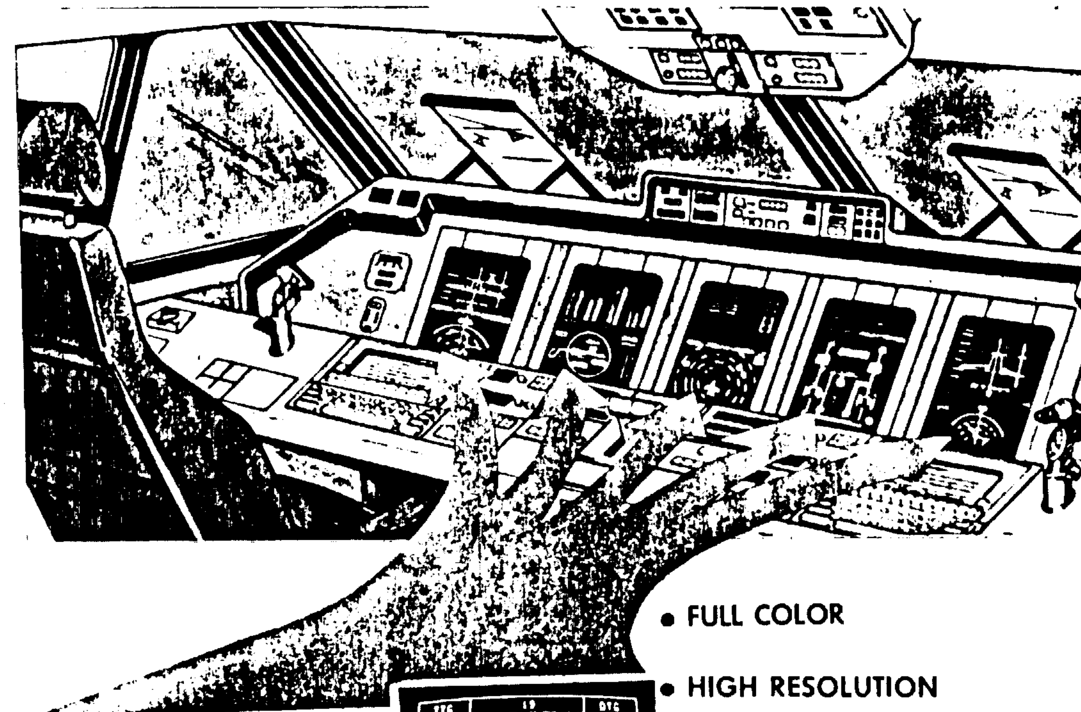


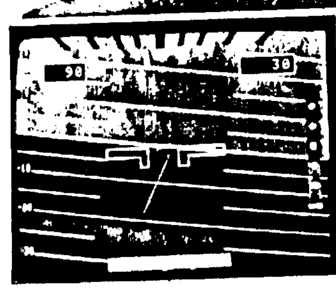
N C STATE- HIGH SPEED GRAPHICS DESIGN

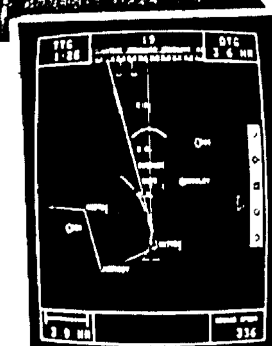


ADVANCED COLORGRAPHIC DISPLAY CAPABILITY

NASA- EXPERIMENTAL PROGRAMMABLE DISPLAY GENERATOR (PDG)







TECHNOLOGY VALIDATION: ADVANCED FLIGHT DISPLAYS

- FULL COLOR
- HIGH RESOLUTION (1024 × 1024 PIXELS)
- RESEARCH FLEXIBILITY
- HIGH-SPEED UPDATE
- 3-D PERSPECTIVE SYMBOLOGY
- SMOOTH ROLLED SYMBOLOGY
- LARGE-AREA SHADING
- FLAT-PANEL COMPATIBLE

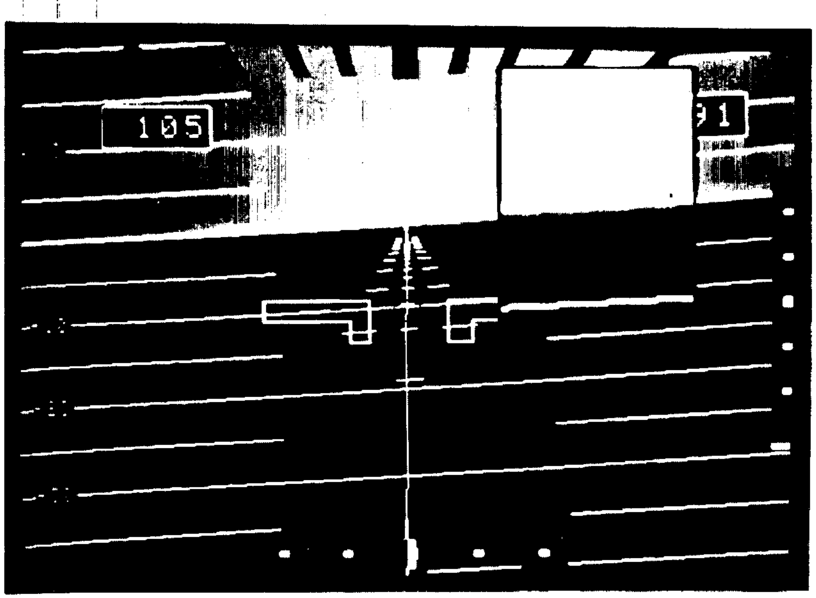
RASTERGRAPHIC DISPLAY GENERATION TECHNOLOGY

RECENT ACCOMPLISHMENTS

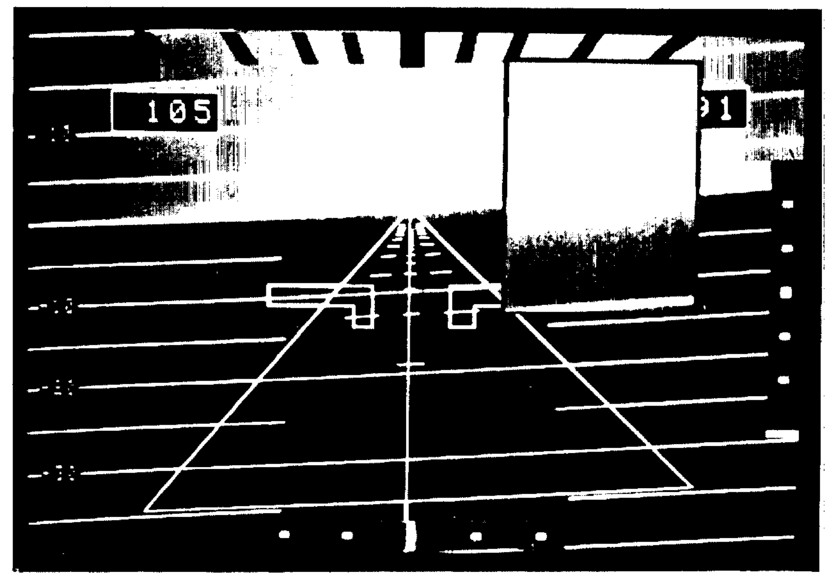
- o REAL-TIME ANTIALIASING ALGORITHMS
- o WINDOW MANAGEMENT OPERATOR
- o PSEUDO 3-D SYMBOLOGY IN REAL-TIME
- o APPLICATION OF SHADED POLYGON CO-PROCESSOR

ADVANCED ALL RASTER PDG PERFORMANCE

CAPABILITY FOR REAL-TIME SMOOTHING OF ROLLED SYMBOLOGY

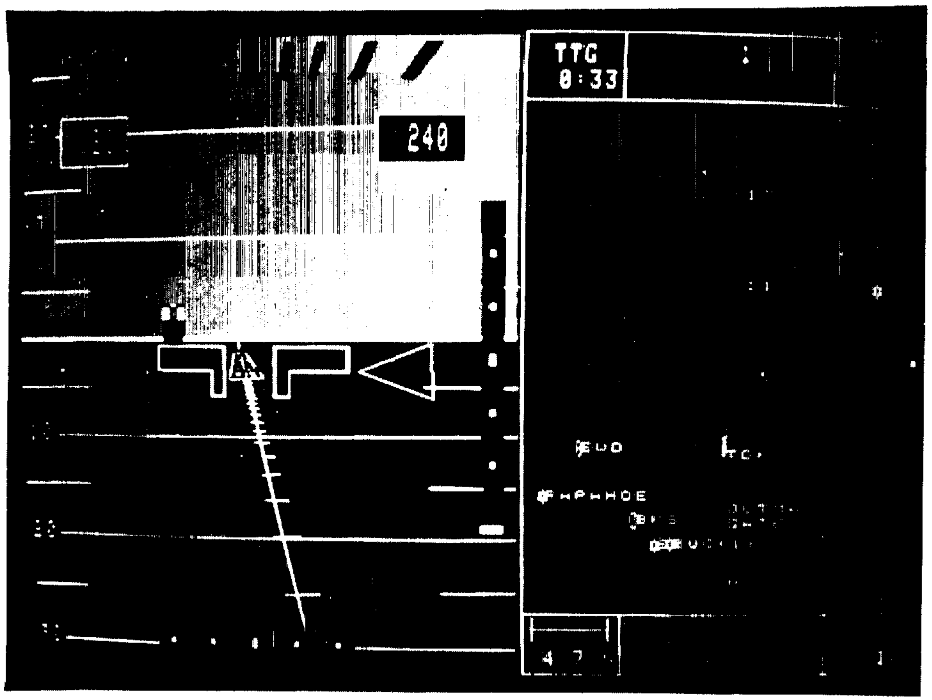


ALIASED

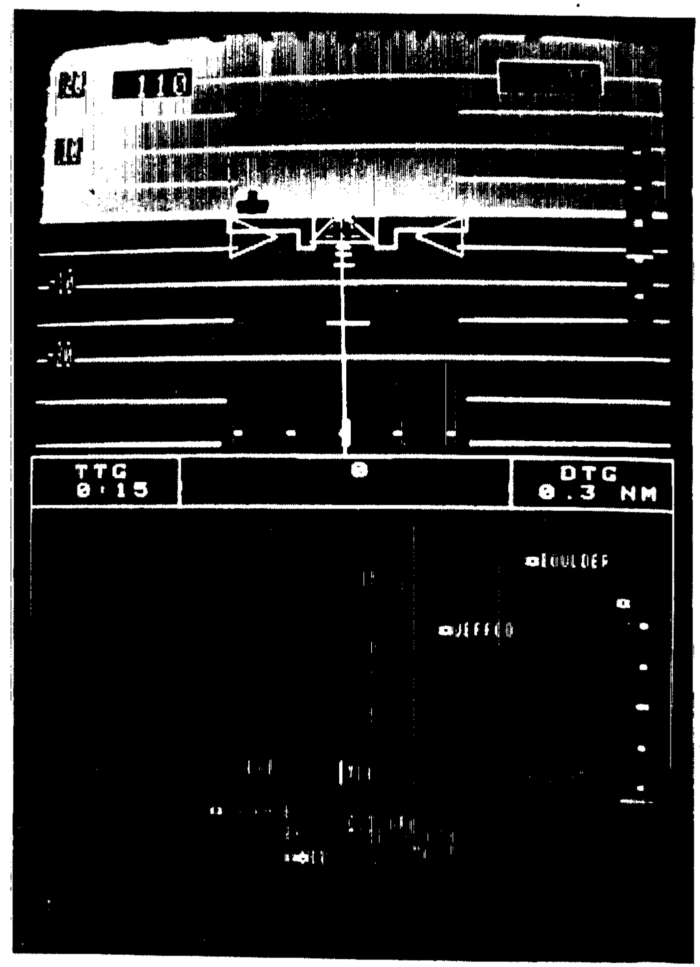


ANTIALIASED

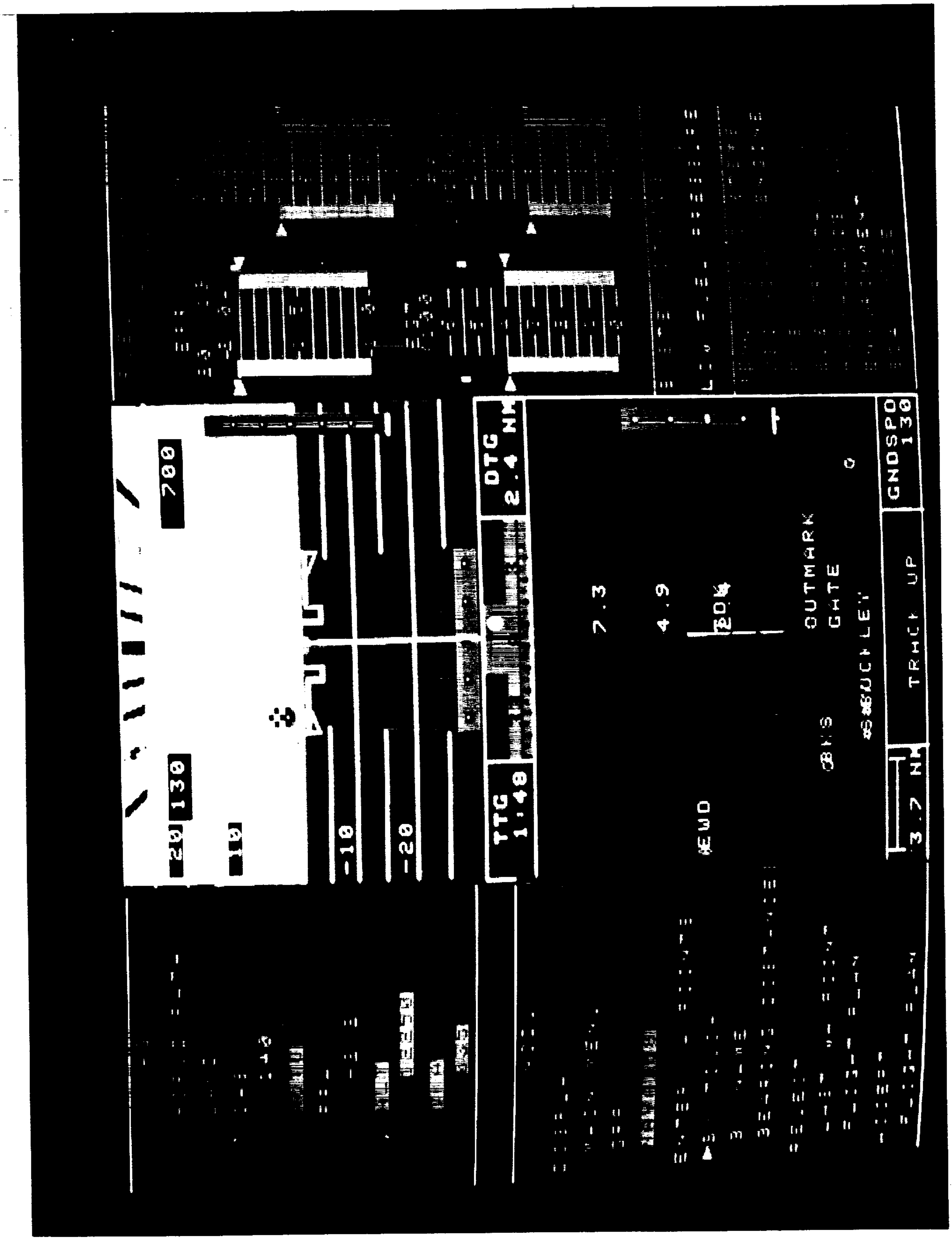
ADVANCED "WINDOW" OPERATOR DEVELOPED FOR INTEGRATED, LARGE-SCREEN FLIGHT DISPLAYS



HORIZONTAL ORIENTATION



VERTICAL ORIENTATION

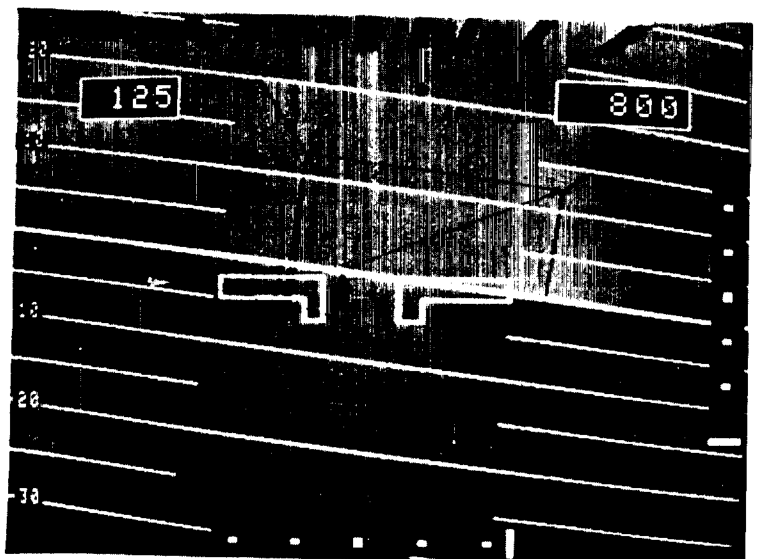


700
130
-10
-20
TTG
DTG
2.4 NM
7.3
4.9
OUTMARK
GATE
TRACK UP
3.7 NM
GNDSPD
130

PSEUDO 3-D SCENE GENERATION

FEATURES MERGED WITH EADI

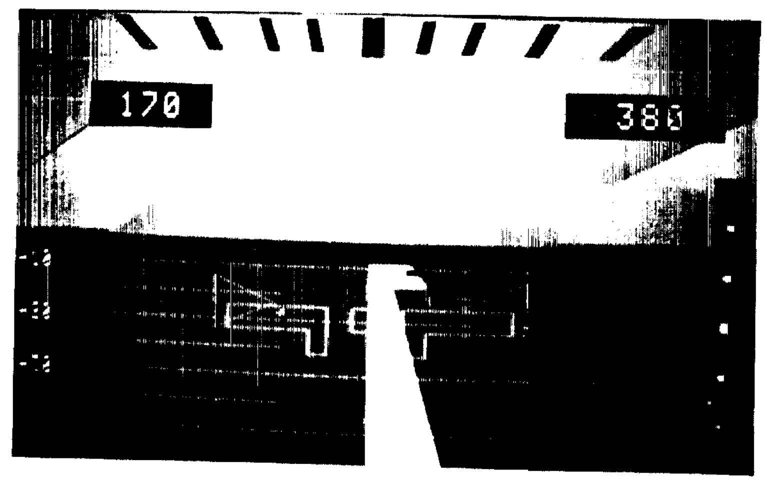
"TUNNEL IN SKY"
PLUS RUNWAY



ADAGE 3000 WITH
HARDWARE FILL

- 512 X 512 RESOLUTION
- .25 MILLION PIXELS/SECOND

"REAL-WORLD" TERRAIN
DATA BASE



ADAGE 3000 WITH
APEX CO-PROCESSOR

- 1024 X 1024 RESOLUTION
- 40 MILLION PIXELS/SECOND

ADVANCED CONTROL/DISPLAY TECHNOLOGY

RECOMMENDATIONS/TECHNOLOGY READINESS

RASTERGRAPHIC DISPLAY GENERATION TECHNOLOGY

- RASTERGRAPHICS IS TO BE PREFERRED TO STROKE TECHNOLOGY FOR THE R&PO APPLICATION BECAUSE OF THE FOLLOWING ADVANTAGES:
  - PRIORTIZED SYMBOLOGY
  - SMALL & LARGE AREA SHADING
  - REDUCED POWER DISSIPATION (CRT DISPLAY)
  - FLAT-PANEL COMPATIBILITY
- THERE EXISTS A VIRTUAL EXPLOSION IN HIGH-PERFORMANCE RASTERGRAPHICS TECHNOLOGY IN THE MARKET PLACE.
- BIT-MAPPED RASTERGRAPHICS SYSTEMS WILL BE THE PREFERRED TYPE FOR R&PO TO PROVIDE PHOTOGRAPHIC-LIKE RESOLUTION/CLARITY.
- A SYSTEMS ARCHITECTURE EMPLOYING PARALLEL PIPELINED PROCESSING AND HARDWARE AUGMENTATION OF GEOMETRIC CALCULATIONS/TRANSFORMATIONS WILL BE NECESSARY FOR REAL-TIME ANIMATION.
- THE GRAPHICS PROCESSOR, WHICH DRIVES WORKSTATION DISPLAYS, SHOULD HAVE FLOATING POINT CAPABILITY TO ENSURE ADEQUATE DYNAMIC SCALING (RANGE) FOR "REAL-WORLD" DISPLAYS.
- NEED FOR A STANDARD HIGH LEVEL LANGUAGE INTERFACE TO A STANDARDIZED GRAPHICS KERNEL EXISTS, FOR REDUCTION OF SOFTWARE DEVELOPMENT COSTS.

RECOMMENDATIONS/TECHNOLOGY READINESS

RASTERGRAPHIC DISPLAY GENERATION TECHNOLOGY (CONT.)

- RESEARCHERS ARE BUSY DEVELOPING/IMPLEMENTING GRAPHICS ALGORITHMS IN HARDWARE USING LSI AND VLSI; THIS REDUCES POWER CONSUMPTION AND VOLUME OF ELECTRONICS AND MAKES PROGRAMMING SIMPLER.

- SEVERAL HIGH-PERFORMANCE RASTERGRAPHICS COMPANIES HAVE BEEN SUCCESSFUL IN DEVELOPING VLSI VERSIONS OF THEIR "GRAPHICS ENGINES" LEADING TO EXTREMELY COMPACT MACHINES WITH ULTRA HIGH REAL-TIME PERFORMANCE IN THE GENERATION OF PSEUDO 3-D DISPLAYS.

- THE STATE-OF-THE-ART IN RASTERGRAPHICS TECHNOLOGY IS DESIRABLE FOR PSEUDO 3-D AND/OR STEREO 3-D APPLICATIONS:
  - RESOLUTION: 1024 X 1280 PIXELS, NONINTERLACED
  - COLOR CRT PIXEL PITCH: .2 MM BETWEEN COLOR TRIADS
  - CONVERGENCE: AUTOMATIC DIGITAL CONVERGENCE
  - VIDEO BANDWIDTH: 100 MHZ
  - PIXEL REFRESH: 60 HZ, MINIMUM

- RUGGEDIZED/MILITARIZED RASTERGRAPHIC TECHNOLOGY IS UNDER DEVELOPMENT: EXAMPLE OF SYSTEM TO BE AVAILABLE BY YEAR'S END:
  - SIZE/WEIGHT: .57 CU FT./42 LBS.
  - RESOLUTION: 1024 X 1024 PIXELS
  - OUTPUT CHANNELS: 2 R-G-B
  - POWER: 450 WATTS

  TECHNOLOGY READINESS LEVEL: 4

DISPLAY MEDIA/INFORMATION MANAGEMENT TECHNOLOGIES

RUSSELL V. PARRISH

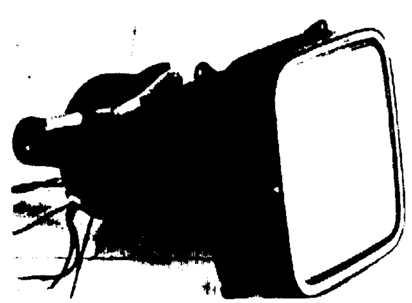
NASA-LANGLEY RESEARCH CENTER

DISPLAY MEDIA RESEARCH: PLANS AND OBJECTIVES

- PERFORM RESEARCH AND DEVELOPMENT ON FLAT-PANEL DISPLAY TECHNOLOGIES WITH POTENTIAL FOR THE FOLLOWING:
  - REPLACEMENT OF THE BULKY CRT FOR LARGE-AREA DISPLAYS
  - INCORPORATION OF NEW MEDIA IN MULTIFUNCTION DEVICES
- PROVIDE NEW MEDIA TECHNOLOGY WHICH CAN OVERCOME MAJOR DISADVANTAGES OF CRT DISPLAY TECHNOLOGY AND PROVIDE SIGNIFICANT NEW ADVANTAGES
  - SHALLOW DEPTH
  - ULTRA HIGH MTBF
  - GRACEFUL DEGRADATION
  - REDUCED POWER
- OVERCOME PRESENT DISADVANTAGES OF FLAT-PANEL TECHNOLOGIES
  - LUMINOUS EFFICIENCY
  - LACK OF COLOR
  - MATRIX ADDRESSING COMPLEXITY
  - HIGH COST
- PERFORMANCE OBJECTIVES:
  - 100 PIXELS/IN
  - NIGHT/DAY VIEWING
  - COLOR CAPABILITY
  - UNIFORM
  - ENVIRONMENTALLY TOLERANT
  - AFFORDABLE

CRT AND FLAT PANEL DISPLAY TECHNOLOGIES COMPARED

CRT



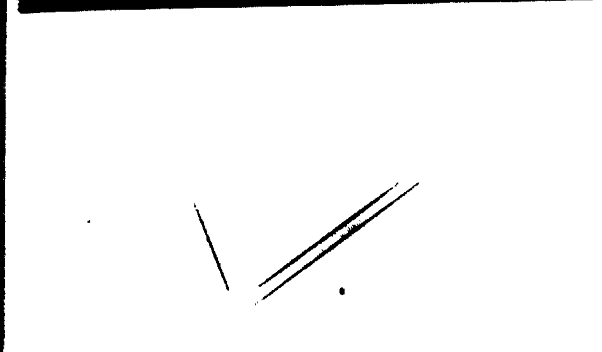
ADVANTAGES

- COST
- BRILLIANCE
- RESOLUTION
- COLOR CAPABILITY
- EASY ADDRESSING
- FAST DYNAMIC RESPONSE
- ENVIRONMENTAL TOLORANCE
- HIGH MTBF

DISADVANTAGES

- HIGH VOLTAGE
- DEPTH/BULK
- IMPLOSION RISK
- LIMITED LIFE (IN HIGH AMBIENT)
- NOT DIRECTLY DIGITALLY COMPATIBLE
- CATASTROPHIC FAILURE
- POWER CONSUMPTION

TFEL



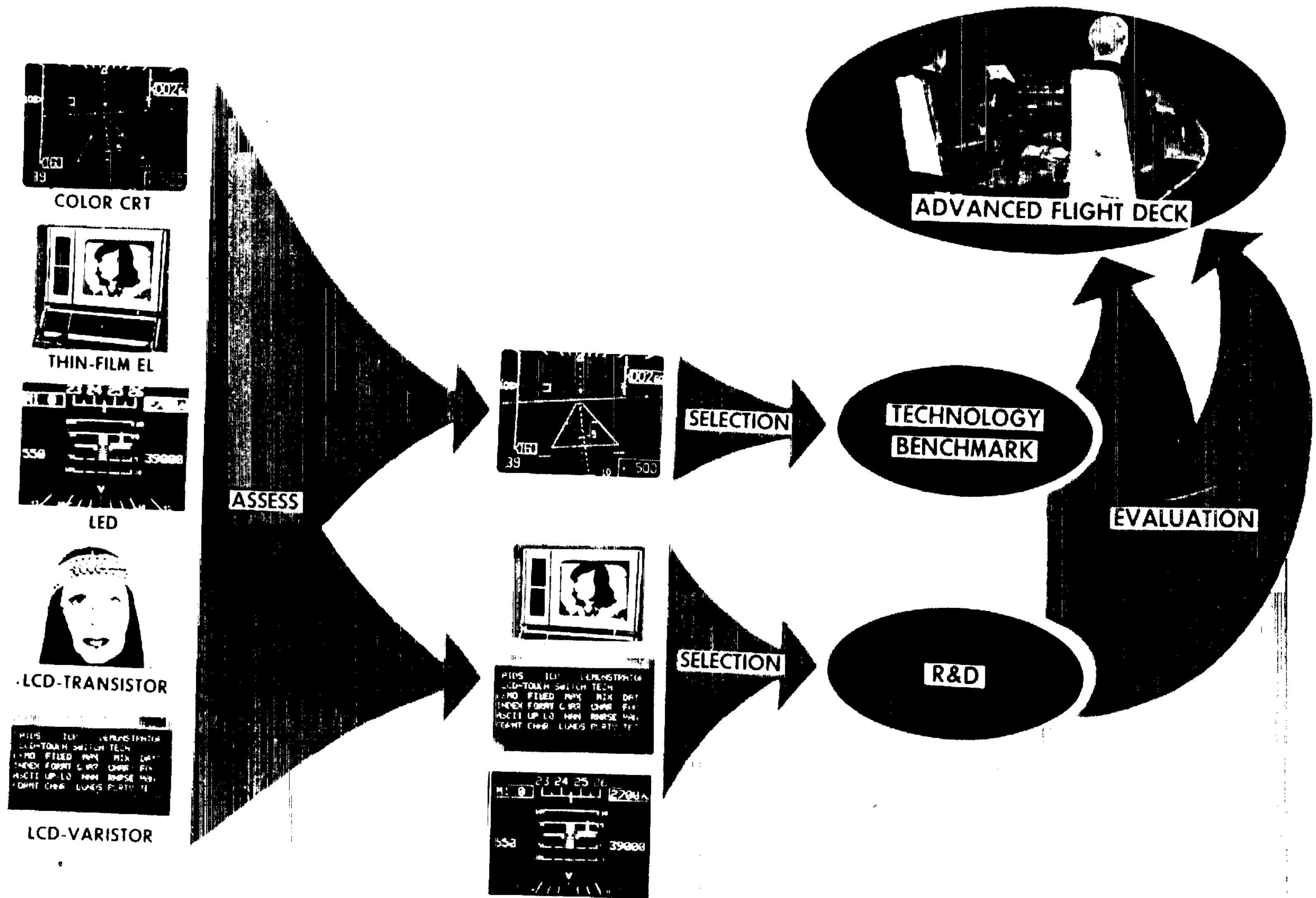
ADVANTAGES

- UNIFORM RESOLUTION
- DIGITALLY COMPATIBLE
- SHALLOW DEPTH
- ULTRA HIGH MTBF
- GRACEFUL DEGRADATION
- REDUCED POWER

DISADVANTAGES

- POOR LUMINOUS EFFICIENCY
- MATRIX ADDRESSING COMPLEXITY
- UNIFORMITY
- LACK OF COLOR
- HIGH COST

COMPARISON AND SELECTION OF ADVANCED ELECTRONIC DISPLAY MEDIA



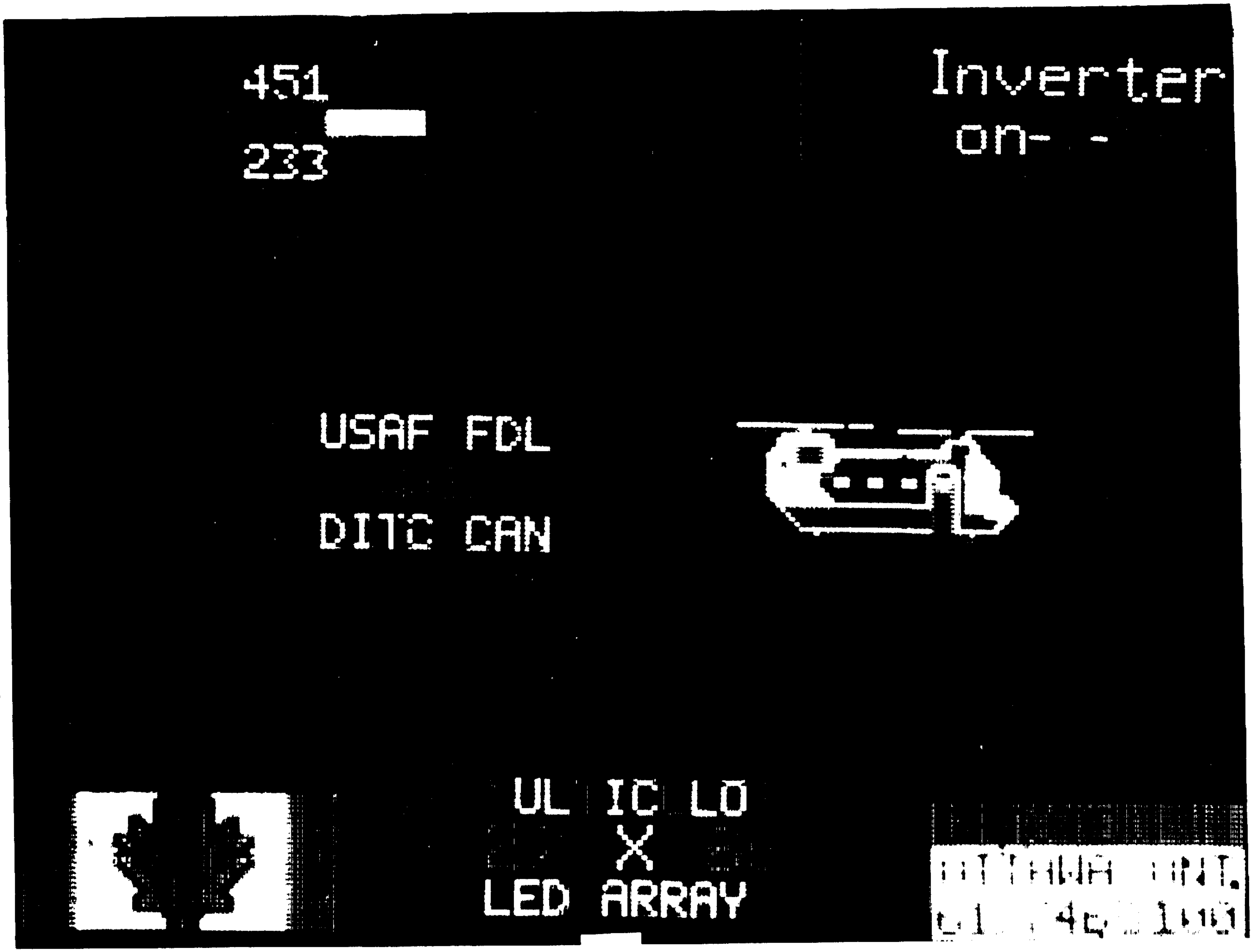
FOUR LEADING CANDIDATES FOR LARGE AREA DISPLAY

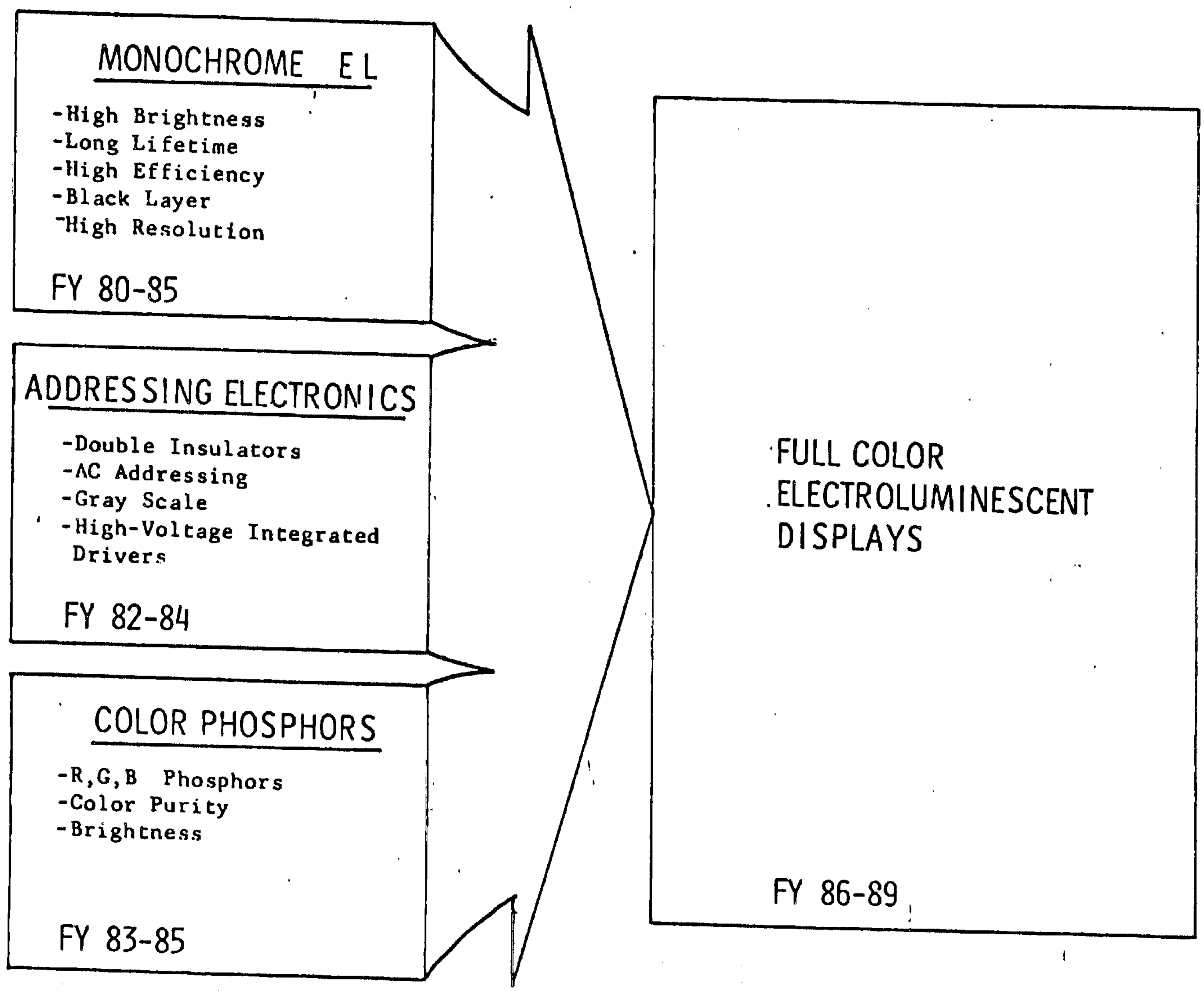
PERFORMANCE CHARACTERISTIC	COLOR CRT	LCD	TFEL	LED.
HIGH RESOLUTION	✓		✓	
LOW POWER CONSUMPTION		✓	✓	
COLOR CAPABILITY	✓			PARTIAL
LOW DEPTH BEHIND PANEL		✓	✓	✓
LOW VOLTAGE OPERATION		✓	✓	✓
VIDEO RATE ADDRESSING	✓	✓	✓	✓
ENVIRONMENTAL TOLERANCE	✓		✓	✓
MEAN-TIME BETWEEN FAILURE			✓	✓
INHERENTLY RUGGED		✓	✓	✓
GRACEFUL DEGRADATION		✓	✓	✓
COST	✓			

THREE-COLOR LIGHT-EMITTING DIODE ARRAY

FOR PROGRAMMABLE DISPLAY SWITCHES

- o NASA FOSTERED PROGRAMMABLE LEGEND LED SWITCH TECHNOLOGY TO DECLUTTER CREWSTATION
  - NOW MARKETED BY BOWMAR, HONEYWELL, LITTON
  - MONOCHROME DISPLAYS ONLY (EITHER RED, YELLOW, OR GREEN)
  - WIDE ACCEPTANCE OF SWITCHES
- o GOAL WAS TO PROVIDE INCREASED RESOLUTION, THREE COLORS, AND UNIFORM BRIGHTNESS
- o INCREASED RESOLUTION AND THREE COLORS HAVE BEEN ACHIEVED BY SUPERIMPOSING MONOLITHIC ARRAYS OF GREEN LED's OVER MONOLITHIC ARRAYS OF RED LED's. WHEN BOTH ARE LIT, EYE PERCEIVES YELLOW.
- o RESEARCH ON UNIFORM BRIGHTNESS, EFFICIENCY, AND PRODUCTION-LINE YIELD CONTINUES


451
233
Inverter
on- -
USAF FDL
DITC CAN
UL IC LO
X
LED ARRAY

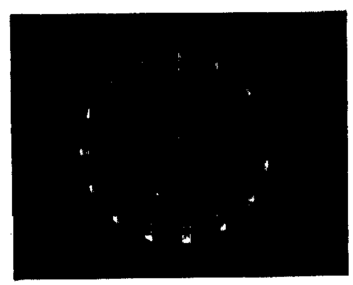

MONOCHROME EL
-High Brightness
-Long Lifetime
-High Efficiency
-Black Layer
-High Resolution
FY 80-85
ADDRESSING ELECTRONICS
-Double Insulators
-AC Addressing
-Gray Scale
-High-Voltage Integrated Drivers
FY 82-84
COLOR PHOSPHORS
-R,G,B Phosphors
-Color Purity
-Brightness
FY 83-85
FULL COLOR ELECTROLUMINESCENT DISPLAYS
FY 86-89

ELECTROLUMINESCENT DISPLAY DEVELOPMENT

MONOCHROME
MEDIUM RESOLUTION

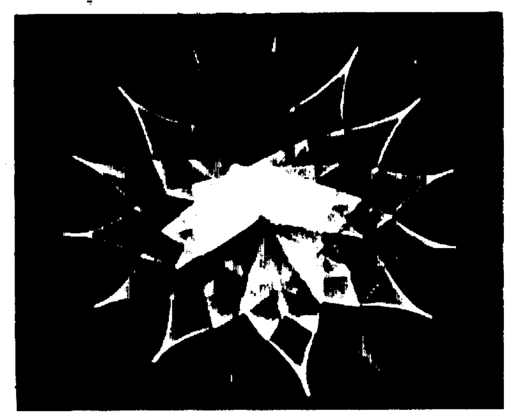
6 INCH DIAGONAL
67 LINES/INCH

MONOCHROME
HIGH RESOLUTION
BLACK LAYER



15 INCH DIAGONAL
100 LINES/INCH
CONTRAST RATIO 2:1 IN SUNLIGHT

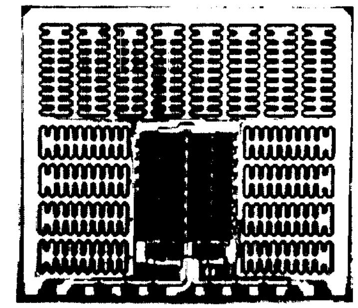
FULL-COLOR
HIGH RESOLUTION
BLACK LAYER



PANORAMIC
100 LINES/INCH
CONTRAST RATIO 4:1 IN SUNLIGHT

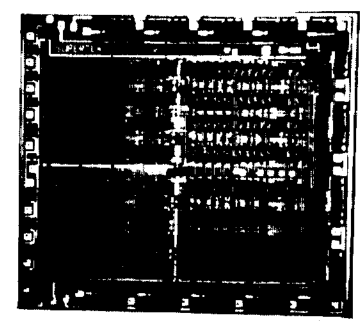
FIRST LABORATORY TESTS OF A HIGH-RESOLUTION LARGE-SCREEN TFEL DISPLAY USING MONOLITHIC DRIVERS

ROW DRIVER



200 VOLTS
16 OUTPUT CHANNELS

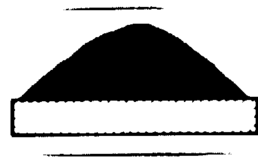
COLUMN DRIVER



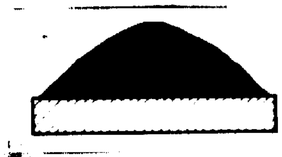
35 TO 60 VOLTS
16 GRAY SHADES
16 OUTPUT CHANNELS

- 100 PIXELS/INCH (512 X 640 PIXELS)
- 8-INCH DIAGONAL SIZE
- RS-170 INTERFACE FOR VIDEO/GRAPHICS
- HIGH-VOLTAGE LSI DISPLAY DRIVERS

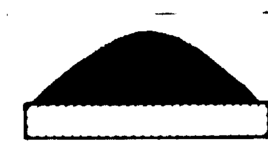
DEVELOPED PHOSPHORS



$ZnS:Mn,TbF_3$

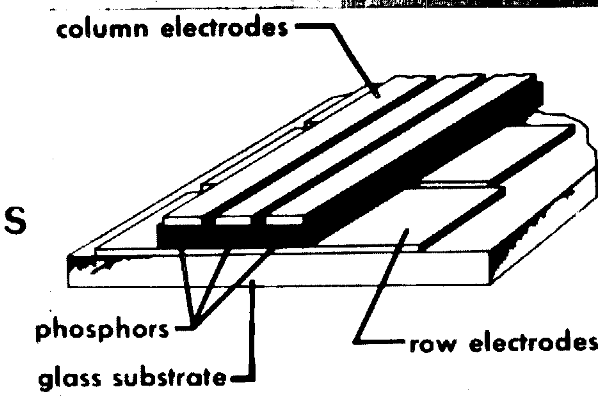


$ZnS:TbF_3$

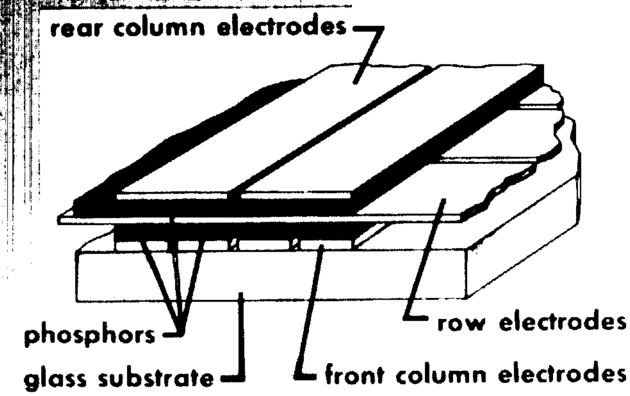


$SrS:CeF_3$

DEVICE CONFIGURATIONS



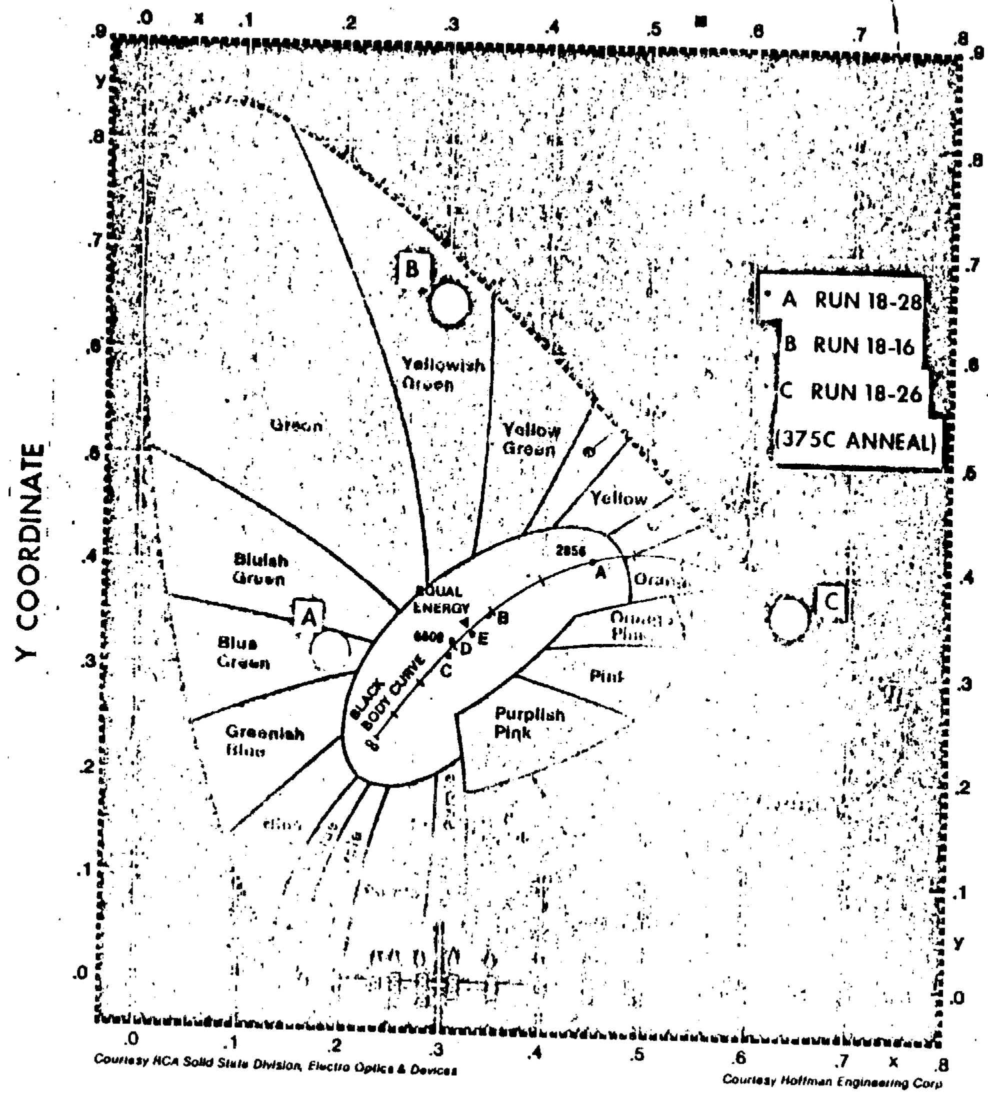
SINGLE LAYER



DOUBLE LAYER

FULL COLOR ELECTROLUMINESCENT DISPLAYS

EMISSION COLORS OF BLUE, GREEN, AND RED TFEL PHOSPHORS



X COORDINATE

KELLY CHART OF COLOR DESIGNATION FOR LIGHTS

* NOTE : RUN 18-28 is $SrS:CeF_3$
RUN 18-16 is $ZnS:TbF_3$
RUN 18-26 is $ZnS:Mn, TbF_3$

ADVANCED CONTROL/DISPLAY TECHNOLOGY

RECOMMENDATIONS/TECHNOLOGY READINESS

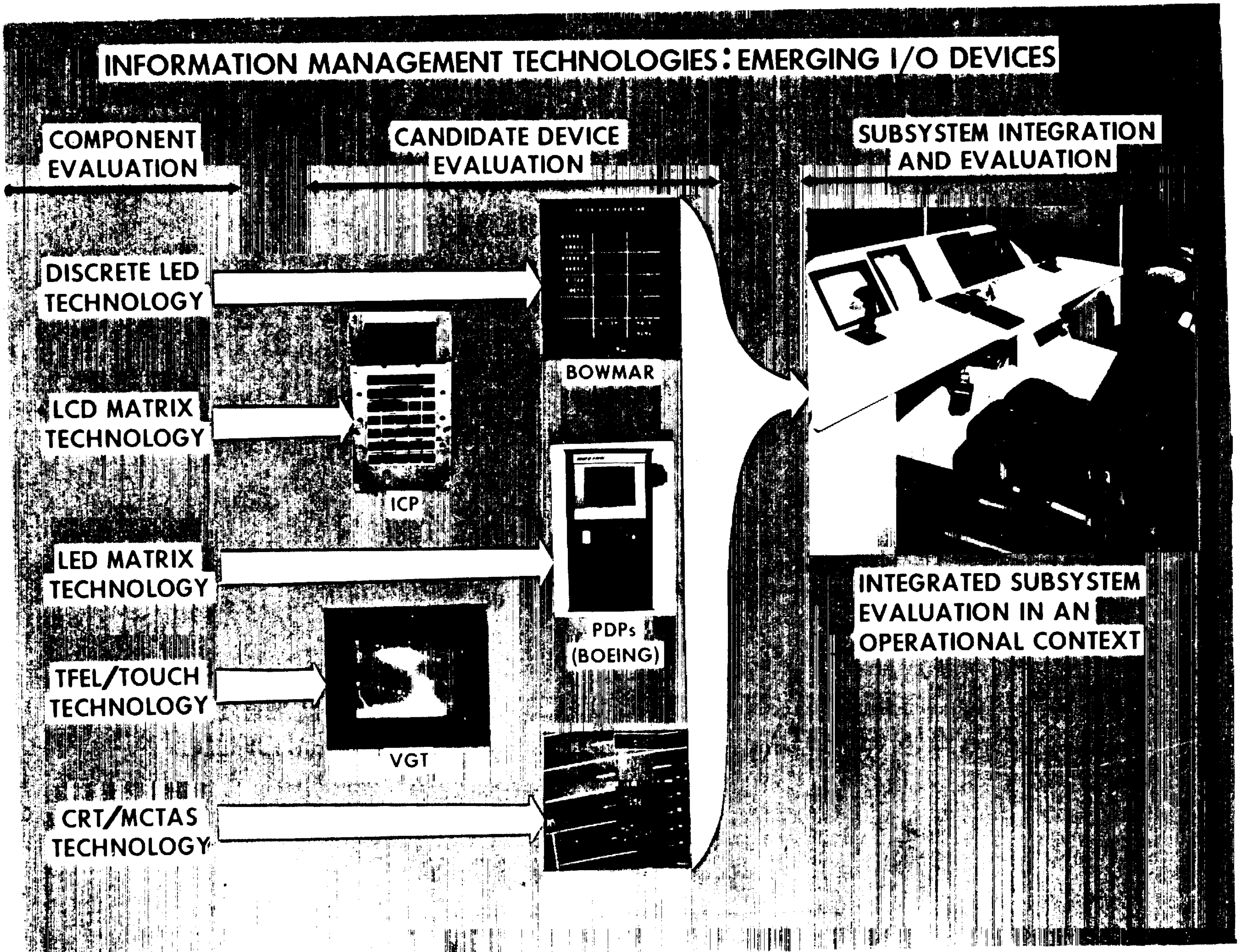
ADVANCED DISPLAY MEDIA

- RUGGEDIZED CRT ONLY MEDIA APPROPRIATE IN NEAR-TERM FOR COLOR DISPLAYS
- LED DISPLAYS CAN PROVIDE TRI-COLOR CAPABILITY, BUT SHOULD BE RESTRICTED TO SMALL-AREAS BECAUSE OF POWER CONSUMPTION
- TFEL TECHNOLOGY A STRONG CONTENDER FOR REPLACING CRT FOR LARGE-AREA DISPLAYS, WITH LCD TECHNOLOGY A CLOSE RUNNER-UP
- TFEL TECHNOLOGY HAS ADVANTAGE IN SIZE: EXPERIMENTAL 16" PANEL EXISTS
- BOTH TFEL & LCD TECHNOLOGY HAVE MECHANISMS FOR PRESERVING CONTRAST WHEN USE NEAR EXTERNAL WINDOWS
  - TFEL: BLACK LAYER
  - LCD: TRANSFLECTIVE MODES
- BOTH TFEL & LCD TECHNOLOGY HAVE PROMISING COLOR MECHANISMS
  - TFEL: PRIMARY COLOR PHOSPHORS (RED & BLUE STILL WEAK)
  - LCD: PRIMARY COLOR FILTERS (GATED BY POLYSILICON TRANSISTORS)
- TFEL TECHNOLOGY IS A SIMPLER, LOWER-COST, HIGHER-YIELD TECHNOLOGY THAT DOES NOT REQUIRE ACTIVE-ELEMENTS WITHIN THE PANEL
- TFEL TECHNOLOGY IS ENVIRONMENTALLY TOLERANT, WHILE LCD TECHNOLOGY REQUIRES HEATERS TO MEET LOWER TEMPERATURE CONDITIONS
- TFEL TECHNOLOGY HAS BEEN FLOWN ON-BOARD THE SPACE TRANSPORTATION SYSTEM (STS) IN THE FORM OF A GRID-SYSTEMS PERSONAL COMPUTER

TECHNOLOGY READINESS LEVEL: 3

INFORMATION MANAGEMENT TECHNOLOGY: PLANS AND OBJECTIVES

- o SERVE AS EVALUATION MECHANISM BY INTEGRATING EMERGING I/O DEVICES, DISPLAY GENERATION TECHNIQUES, AND INFORMATION MANAGEMENT CONCEPTS IN:
    - LABORATORY BENCH STUDIES
    - SUB-SYSTEM APPLICATION EXPERIMENTS
    - INTEGRATED SYSTEM EXPERIMENTS
- o INTEGRATION TECHNIQUES WILL EMPHASIZE THESE MANAGEMENT CONCEPTS FOR INFORMATION DATABASES AND FUNCTIONS
    - MULTI-PURPOSE, CONSOLIDATED CONTROLS
    - INTEGRATED CONTROLS/DISPLAYS
    - INTELLIGENT AUTOMATION
    - DECISION AIDING STRATEGIES
    - VIDEO DISK STORAGE/RETRIEVAL


INFORMATION MANAGEMENT TECHNOLOGIES: EMERGING I/O DEVICES
COMPONENT EVALUATION
CANDIDATE DEVICE EVALUATION
SUBSYSTEM INTEGRATION AND EVALUATION
DISCRETE LED TECHNOLOGY
LCD MATRIX TECHNOLOGY
LED MATRIX TECHNOLOGY
TFEL/TOUCH TECHNOLOGY
CRT/MCTAS TECHNOLOGY
BOWMAR
ICP
PDPs (BOEING)
VGT
INTEGRATED SUBSYSTEM EVALUATION IN AN OPERATIONAL CONTEXT

PILOT/AUTOPILOT INTERFACE USING HANDS-ON

THROTTLE AND STICK CONCEPT

- o INTERACTIVE HIGH-SPEED GRAPHICS CAPABILITY INTEGRATED WITH MULTIFUNCTION CONTROL DEVICES TO REDUCE COCKPIT CLUTTER.

MULTIFUNCTION CONTROLS USAGE:

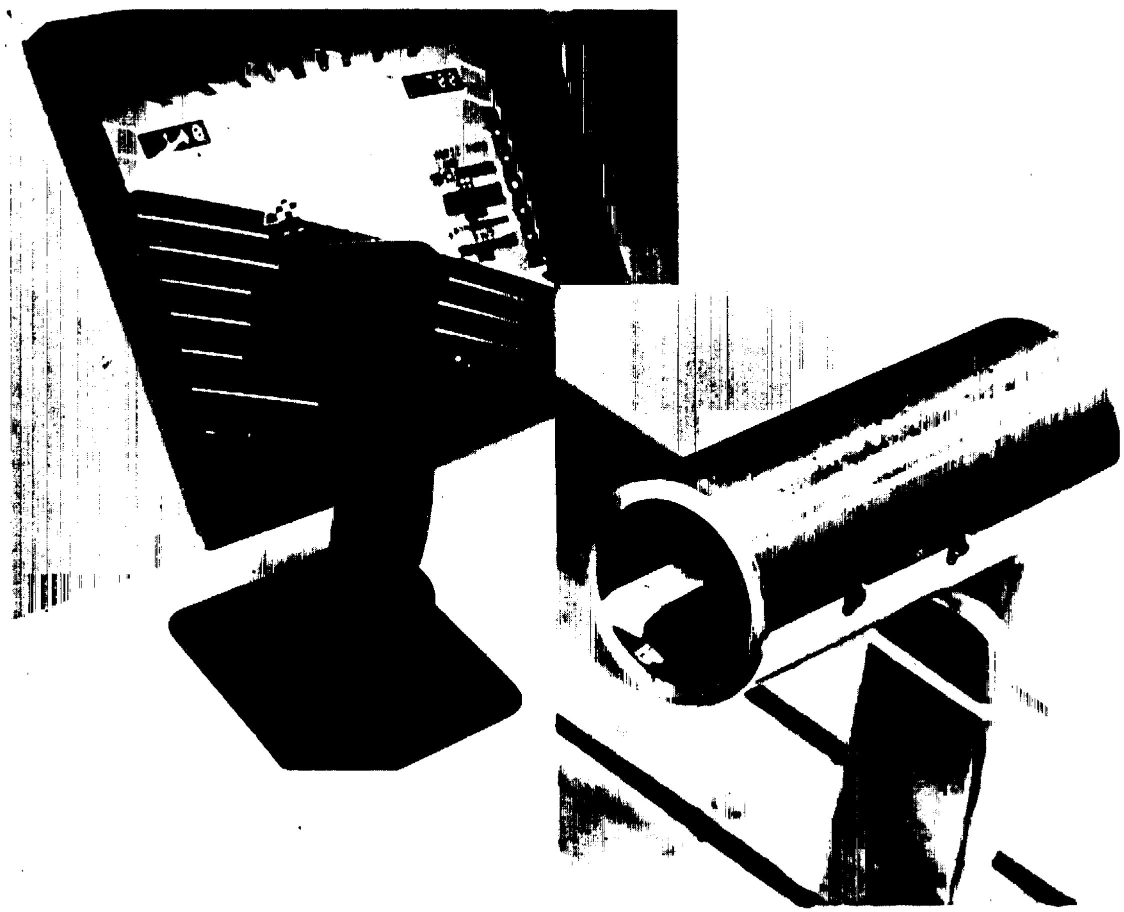
- CURSOR MOVEMENT
- MODE SELECTION
- NUMERICAL CHANGES TO FLIGHT PARAMETERS

INTERACTIVE GRAPHICS USAGE:

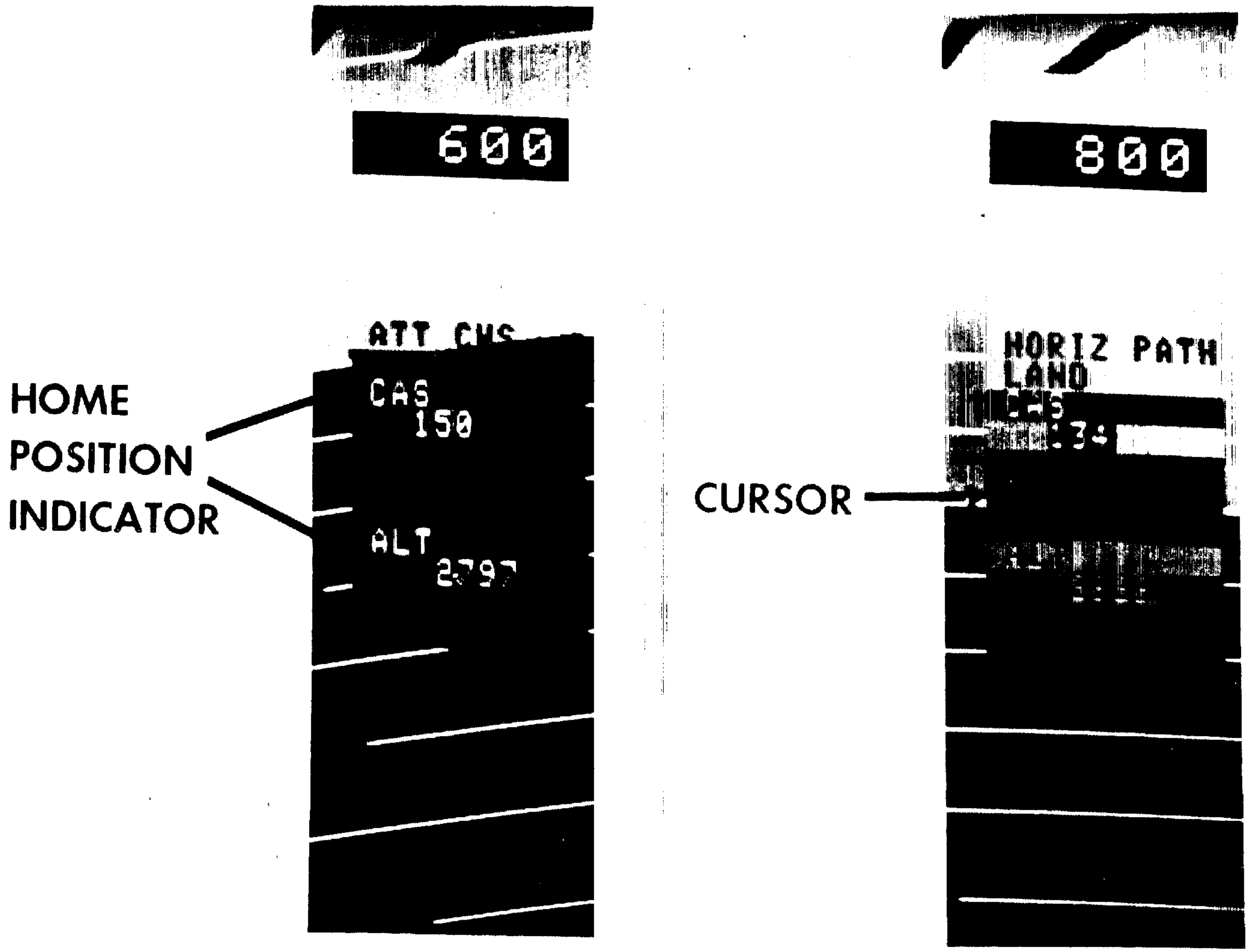
- MENU GENERATION
- MODE SELECTION COLOR CHANGES
- FLIGHT VARIABLE DISPLAYS

- o SIMULATOR EVALUATION COMPARING CONVENTIONAL AUTOPILOT INTERFACE TO INTEGRATED CONCEPT WILL FOLLOW. INITIAL PILOT REACTION FAVORABLE.

MULTIFUNCTION CONTROL THROTTLE & STICK PILOT/AUTOPILOT INTERFACE



MODE-BASED MENU DISPLAYS



<u>ADVANCED NAVIGATIONAL INTERFACE FOR</u>

<u>MODERN TRANSPORT AIRCRAFT</u>

- <u>OBJECTIVE</u>: INVESTIGATE USE OF ADVANCED DISPLAY MEDIA TO SIMPLIFY PILOT's FLIGHT PLAN MANAGEMENT FUNCTIONS

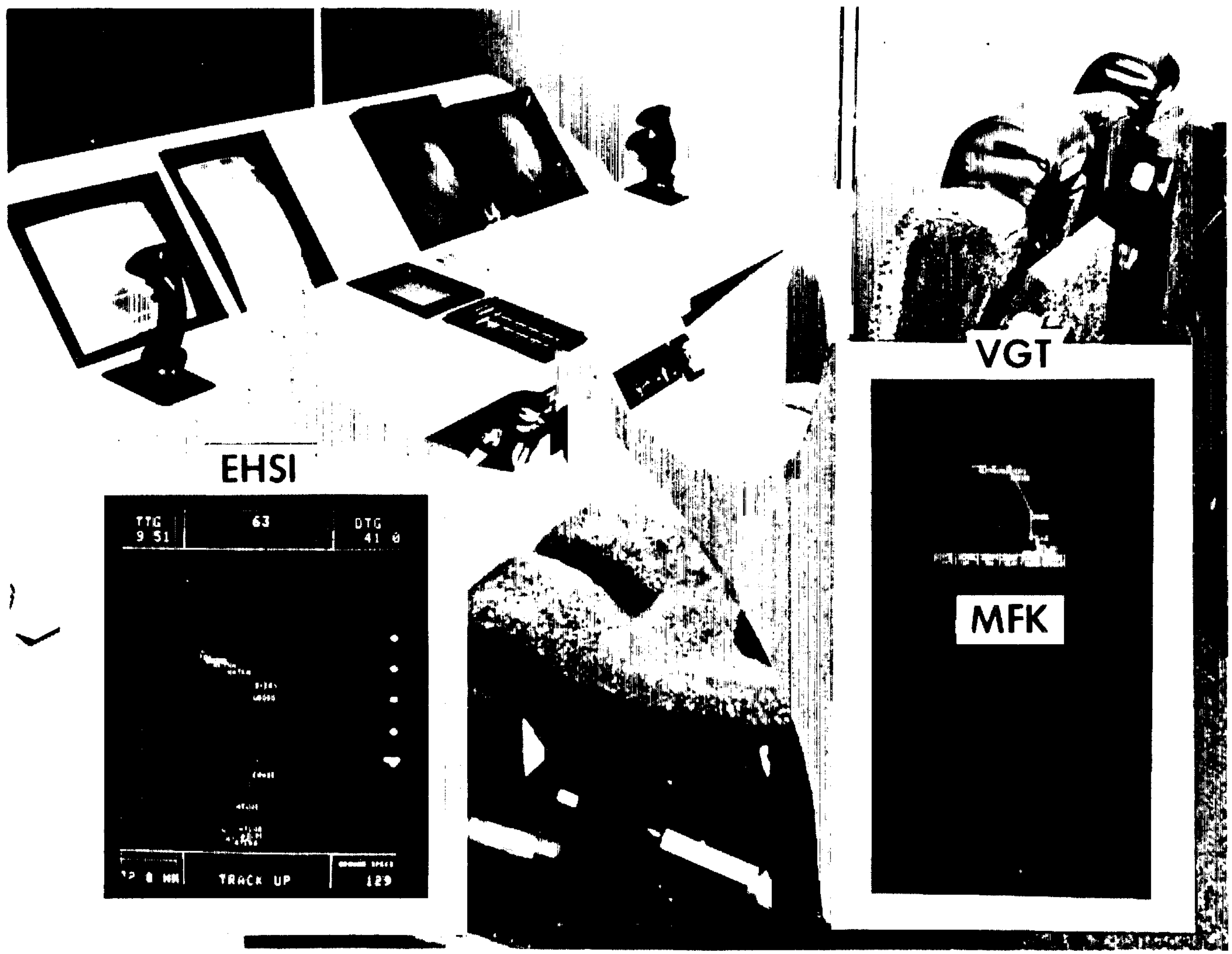
- <u>APPROACH</u>: UTILIZE HIGH-SPEED RASTER GRAPHICS DISPLAY GENERATOR AND AN ADVANCED MEDIA NAVIGATION CONTROL DISPLAY UNIT (NCDU), CONSISTING OF FLAT PANEL DISPLAY (VGT) AND MULTIFUNCTION LED KEYBOARD (MFK)

  - VGT AND MFK USED TO MODIFY AND DISPLAY FLIGHT PLAN REVISIONS PRIOR TO ACCEPTANCE AND UPDATE OF EHSI ACTIVE FLIGHT PLAN
  - MODIFICATIONS ENTERED BY EITHER TOUCH OR KEYBOARD

- <u>RESULTS</u>: PRELIMINARY PILOT EVALUATION IN SIMULATED "ATLANTA TO DENVER" FLIGHT VALIDATES OVERALL CONCEPT AND ADVANCED MEDIA DEVICE ACCEPTANCE

- <u>FUTURE</u>: TO PERFORM DETAILED EXPERIMENTAL EVALUATIONS AND FURTHER REFINE NAVIGATIONAL SYSTEM

ADVANCED NAVIGATIONAL INTERFACE FOR MODERN TRANSPORT AIRCRAFT



ADVANCED CONTROL/DISPLAY TECHNOLOGY

RECOMMENDATIONS/TECHNOLOGY READINESS

INFORMATION MANAGEMENT TECHNOLOGY

- MULTIFUNCTION CONTROL INTERFACES MATURING RAPIDLY, THROUGH A MERGING OF ADVANCED DISPLAY MEDIA AND MICROPROCESSOR TECHNOLOGY, TO PROVIDE:
  - HIGHLY RELIABLE I/O
  - CONTROLS CONSOLIDATION/DECLUTTER
  - REDUCED TRAINING/CROSS-TRAINING
  - REDUCED WEIGHT, COMPLEXITY, COST
- INTEGRATION OF CONTROLS/DISPLAYS EVOLVING FROM DUPLICATION OF TRADITIONAL IMPLEMENTATIONS TO METHODOLOGIES THAT REDUCE WORKLOAD BY OPTIMIZING USE OF:
  - INTELLIGENT AUTOMATION
  - DECISION AIDING STRATEGIES
  - INNOVATIVE USES OF TOUCH, VOICE, AND TACTILE I/O
  - HIGHLY INTERACTIVE GRAPHICS
- INFORMATION STORAGE/RETRIEVAL SYSTEMS PROMISING:
  - VIDEO DISK "READ-ONLY" NEARING RUGGEDIZATION
  - VIDEO DISK "WRITE" CAPABILITY STILL DOWNSTREAM

TECHNOLOGY READINESS LEVEL: 2

ADVANCED CONTROL/DISPLAY TECHNOLOGY

SUMMARY/CONCLUSIONS

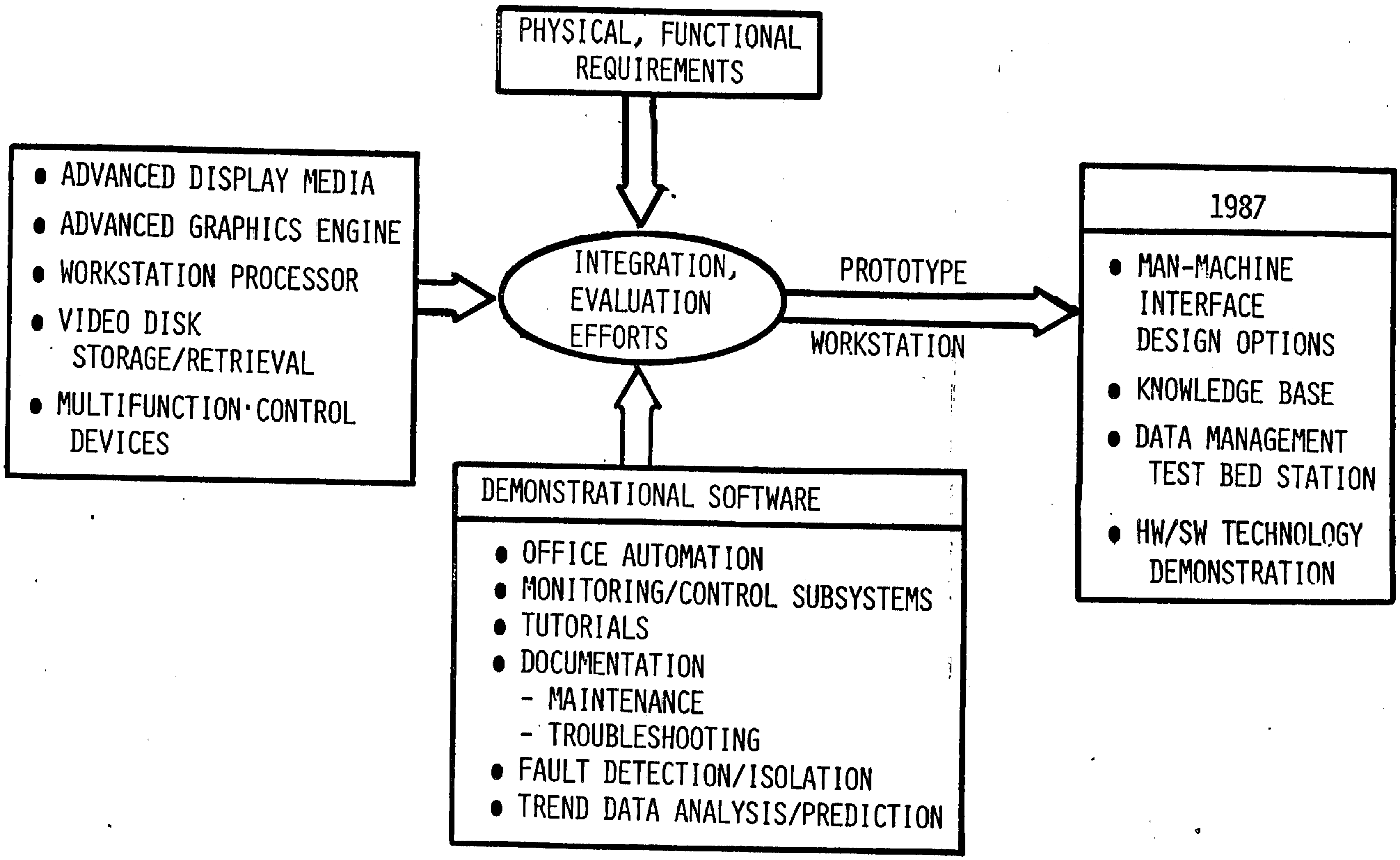
- o IMPORTANT TECHNOLOGIES FOR CONSOLIDATING AUTOMATING, AND INTEGRATING WORKSTATIONS FOR RENDEZVOUS & PROXIMITY OPERATIONS:
  - ADVANCED DISPLAY MEDIA
  - IMAGE/DISPLAY GENERATION
  - IMAGE STORAGE/RETRIEVAL
  - MULTI-FUNCTION CONTROL DEVICES
  - INTEGRATION METHODOLOGIES
- o LARC IS HELPING TO PROVIDE A TECHNOLOGY BASE IN THESE AREAS WITH TWO EFFORTS:
  - R&T BASE: CREW STATION TECHNOLOGY
  - FOCUSED TECHNOLOGY: ELECTRONIC CONTROL/DISPLAY INTERFACE
- o EXCITING RESEARCH RESULTS ARE BEING EXPERIENCED IN R&T BASE EFFORT
  - HIGH-PERFORMANCE, HIGH FLEXIBILITY RASTER DISPLAY GENERATION
  - MONOCHROME & COLOR TFEL FLAT-PANEL DISPLAY MEDIA
  - MONOCHROME & TRI-COLOR LED SMALL-AREA DISPLAY MEDIA
  - CONTROLS CONSOLIDATION USING LCD & LED SWITCHES/KEYBOARDS
  - CREW STATION DECLUTTER USING PICTORIAL GRAPHICS & MULTIFUNCTION CONTROL THROTTLE & STICK (MCTAS)
  - PILOT EVALUATIONS OF ADVANCED WORKSTATION INTERFACES

ADVANCED CONTROL/DISPLAY TECHNOLOGY

SUMMARY/CONCLUSIONS (CONT.)

- o R & T BASE EFFORT HAS ALREADY BEGUN TO HAVE AN IMPACT ON INDUSTRY
  - MAJOR AIRFRAMER SELECTED ADAGE 3000 RASTER TECHNOLOGY FOR ADVANCED SIMULATION
  - NASA ARC & LARC SELECTED ADAGE 3000 TECHNOLOGY FOR ADVANCED CONCEPT SIMULATION
  - ADAGE 3000 TECHNOLOGY SELECTED BY NAVY AND AIRFORCE FOR SYSTEMS INTEGRATION
  - ADAGE 3000 TECHNOLOGY IN USE AT OVER 100 COMPANIES & UNIVERSITIES
  - PROGRAMMABLE LED PUSHBUTTON IS NOW A MAJOR PRODUCT AND IS ESTABLISHING A TREND IN THE INDUSTRY
  - LED-BASED MULTIFUNCTION KEYBOARDS AVAILABLE FROM SEVERAL FIRMS
  - AN EFFICIENT NEW GREEN PHOSPHOR IS AVAILABLE FOR TFEL DISPLAYS
  - LARGER-SIZED, HIGHER-RESOLUTION TFEL DISPLAYS ARE AVAILABLE
  - TUTORIAL ON CREW STATION TECHNOLOGY GIVEN AT 4TH & 5TH DASC
- o FOCUSED TECHNOLOGY EFFORT IS A NEW ONE WHICH WILL DELIVER PRODUCTS IN THE FOLLOWING AREAS:
  - MAN/MACHINE INTERFACE DESIGN OPTIONS
  - KNOWLEDGE BASE
  - WORKSTATION FOR DATA MANAGEMENT TEST BED

FOCUSED TECHNOLOGY: SPACE STATION ELECTRONIC CONTROL/DISPLAY INTERFACE TECHNOLOGY



SYSTEM ENGINEERING APPROACH TO THE ADVANCED WORK STATION FOR SPACE STATION

JOHN HUSSEY
SPACE STATION PROJECT

GRUMMAN

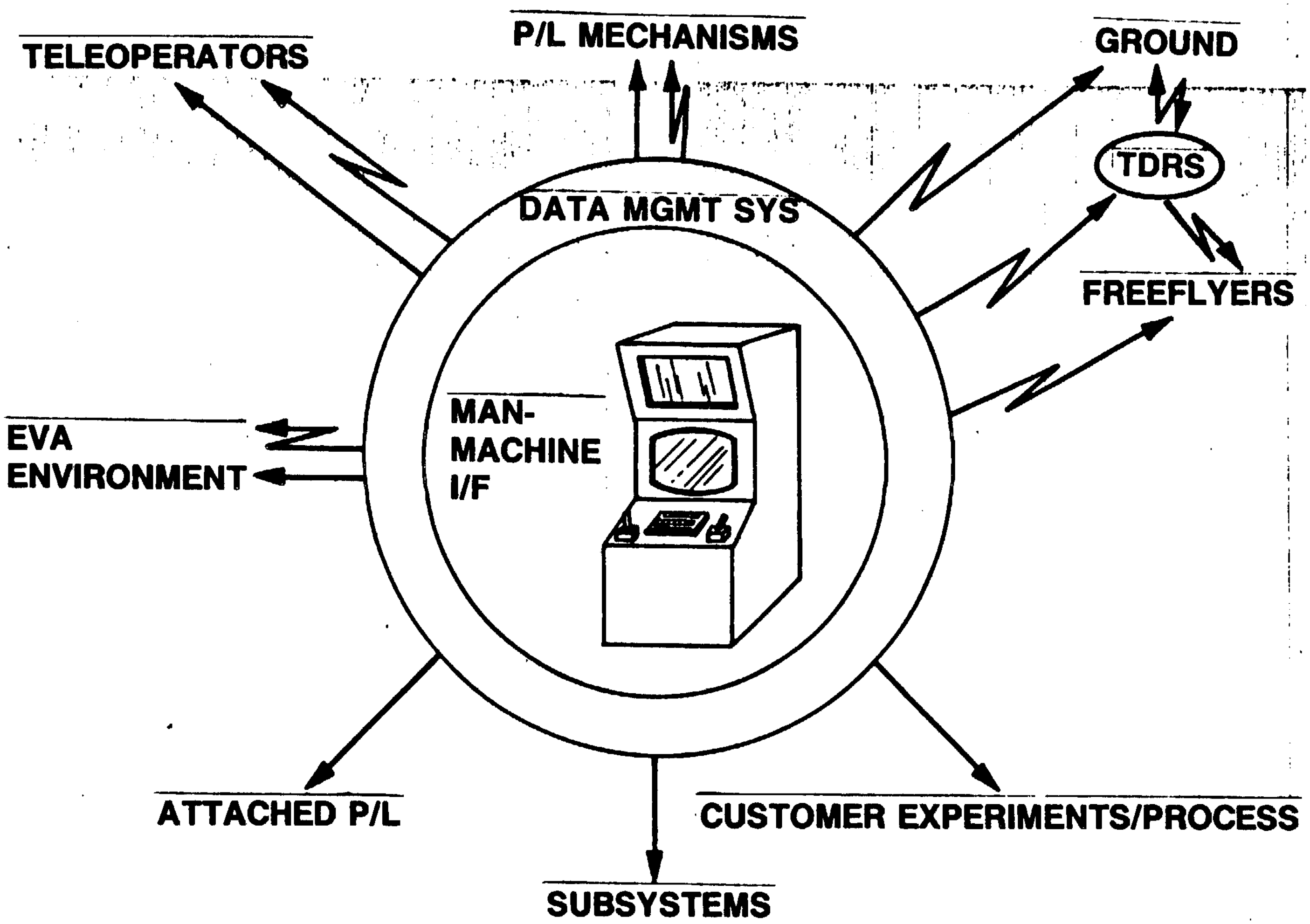
0190-000D

WORK STATION STEERING ELEMENTS

- AUTOMATION
- AUTONOMOUS OPERATION ON ORBIT THAT GROWS TO TOTAL AUTONOMY
  - ROLE OF GROUND CHANGES
- UTILIZATION WITH GROWTH CAPABILITY EXPLOITS TECHNOLOGY FOR PRODUCTIVITY
- SUPPORTS CUSTOMER COMMERCIALIZATION OF SPACE
  - VAGUELY DEFINED AT THIS POINT
- SUPPORTS CONSTRUCTION, MAINTENANCE & OPERATIONS OF THE STATION
  - IVA & EVA
- HUMAN PRODUCTIVITY ELEMENTS THAT INFLUENCE THE FUNCTIONAL MODELING & STRUCTURING OF THE INFORMATION SYSTEM

0190-001D

C & D INFRASTRUCTURE



0190-002D

HUMAN PRODUCTIVITY

- PRODUCTIVITY ASPECTS
- ELEMENT RELATIONSHIPS

STEERING ELEMENTS

- MAN/MACHINE ROLES
- AUTONOMY FROM THE GROUND
- SAFETY
- HUMAN FACTORS ENGINEERING

COMMON ELEMENTS

- ANTHROPOMETRICS
- INFORMATION MANAGEMENT
- COMMUNICATIONS
- INFLIGHT MAINTENANCE
- QUALITY ASSURANCE
- LOGISTICS
- INFLIGHT TRAINING SIMULATORS
- SOCIAL/PSYCHOLOGICAL

CUSTOMER SERVICES
- IVA SYSTEMS
- EVA SYSTEMS

SPACE STATION SUPPORT
- ARCHITECTURE
- CREW FACILITIES
- CREW ACTIVITIES
- GROUND SUPPORT

0190-003D

HUMAN FACTORS ENGINEERING CONSIDERATIONS

- HUMAN PERCEPTION
  - — PHYSICAL & PSYCHOPHYSICAL MEASUREMENTS
  - — PERCEPTUAL PERFORMANCE
  - — ADAPTATION & TRAINING EFFECTS
  - — SOURCES OF PERCEPTUAL ERRORS
- HUMAN PHYSICAL CHARACTERISTICS
  - — MICRO "g" BIOMECHANICS & KINEMATICS
  - — HEALTH/STRESS STATE
  - — SOCIAL/PSYCHOLOGICAL
- HUMAN LEARNING, MEMORY & ATTENTION
  - — RECALL & RECOGNITION CAPABILITIES
  - — SELECTIVE & DIVIDED ATTENTION
  - — MEASURING & PREDICTING MENTAL WORKLOADS
  - — DECISION MAKING AND ORGANIZATION

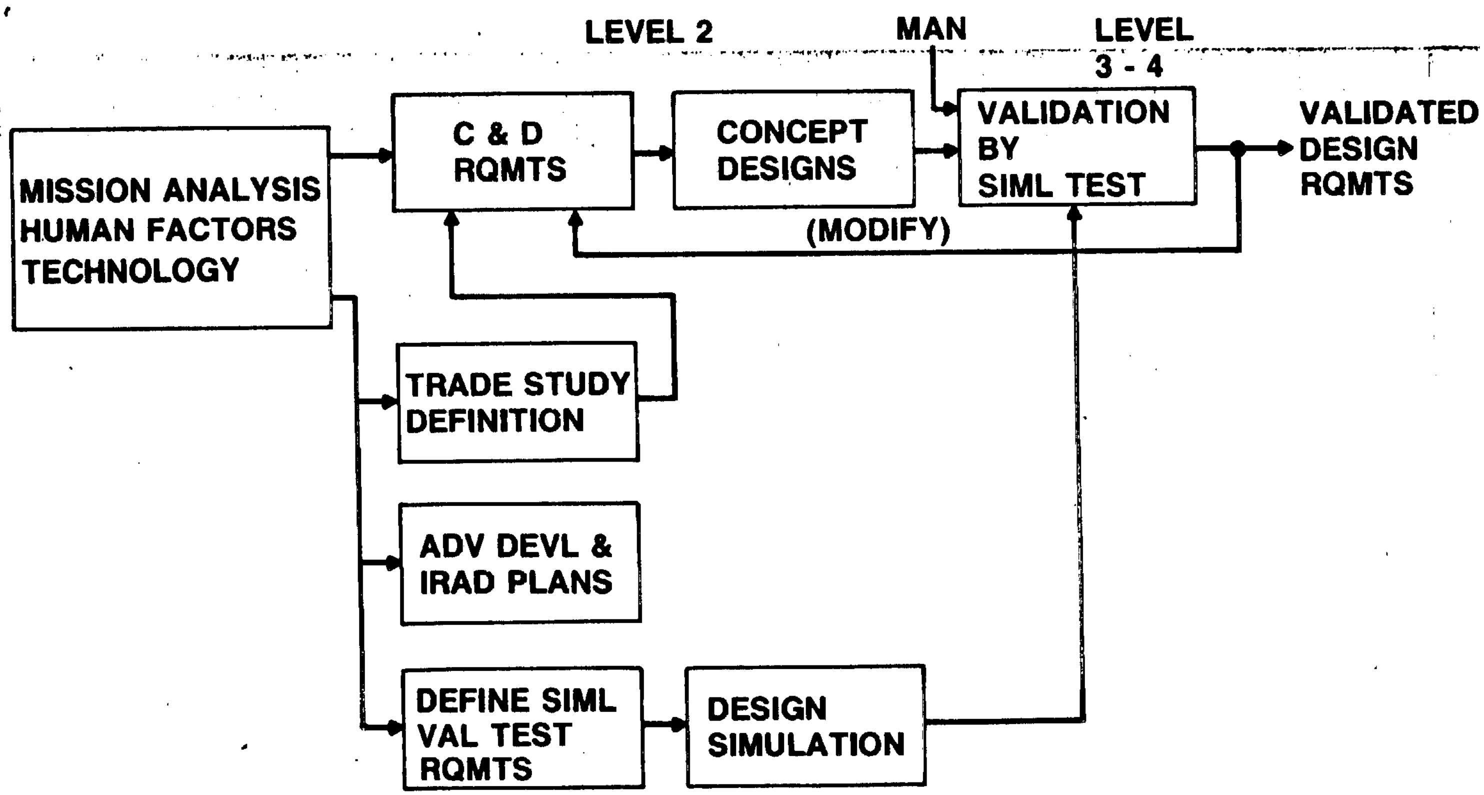
0190-004D

HUMAN FACTORS ENGINEERING CONSIDERATIONS (CONT)

- HUMAN INFORMATION PROCESSING
  - COLOR ENCODING FOR INFORMATION ENHANCEMENT
  - INFORMATION OVERLOADS
  - SOURCES OF COGNITIVE ERRORS
  - ANALOG VS DIGITAL DATA
  - VISIBILITY/ACCESSABILITY (PRIME - SUPORTIVE - BACKUP)
- OPTIMAL MAN/MACHINE INTERFACE & HUMAN TASK ANALYSIS
  - HUMAN VS MACHINE TASK SPECIFICATION
  - SUB-TASK SEGMENTATION, SEQUENCES & DURATIONS
  - SKILL & KNOWLEDGE REQUIREMENTS
  - INTERNATIONAL CREW CONSIDERATIONS
- WORK SPACE CONSIDERATIONS
  - AMBIENT NOISE/LIGHTING
  - REDUCTION OF FATIQUE & STRESS
  - ACCESS & STOWAGE
  - PORTABILITY

0190-004A(D)

PHASE A/B STUDY FLOW



0190-005D

MISSION ANALYSIS FACTORS

- DESIGN REFERENCE MISSIONS
- MISSION FUNCTIONAL ANALYSIS
- TASK ANALYSIS
- TIME LINES & PROCEEDURES
- PRELIMINARY MEASUREMENTS LIST

IDENTIFY GENERIC Functional PATHS AND VALIDATION TEST REQUIREMENTS

0190-006D

DESIGN APPROACH FOR WORK STATION

- GENERIC HARDWARE DESIGN
- KITABLE FOR FUNCTIONAL EXCEPTIONS (I.E. HANDCONTROLLERS)
- AUTONOMOUS USERS NODE
- STANDARD OPERATING SYSTEM SOFTWARE
- FUNCTIONAL APPLICATIONS SOFTWARE
- 'STANDARD MAN MACHINE INTERFACE
  - DISPLAY FORMATS (SOFTWARE)
  - I/O DEVICES (HARDWARE)
- ADAPTABLE ANTHROPOMETRIC DESIGN
  - FUNCTION
  - LOCATION (ARCHITECTURE)

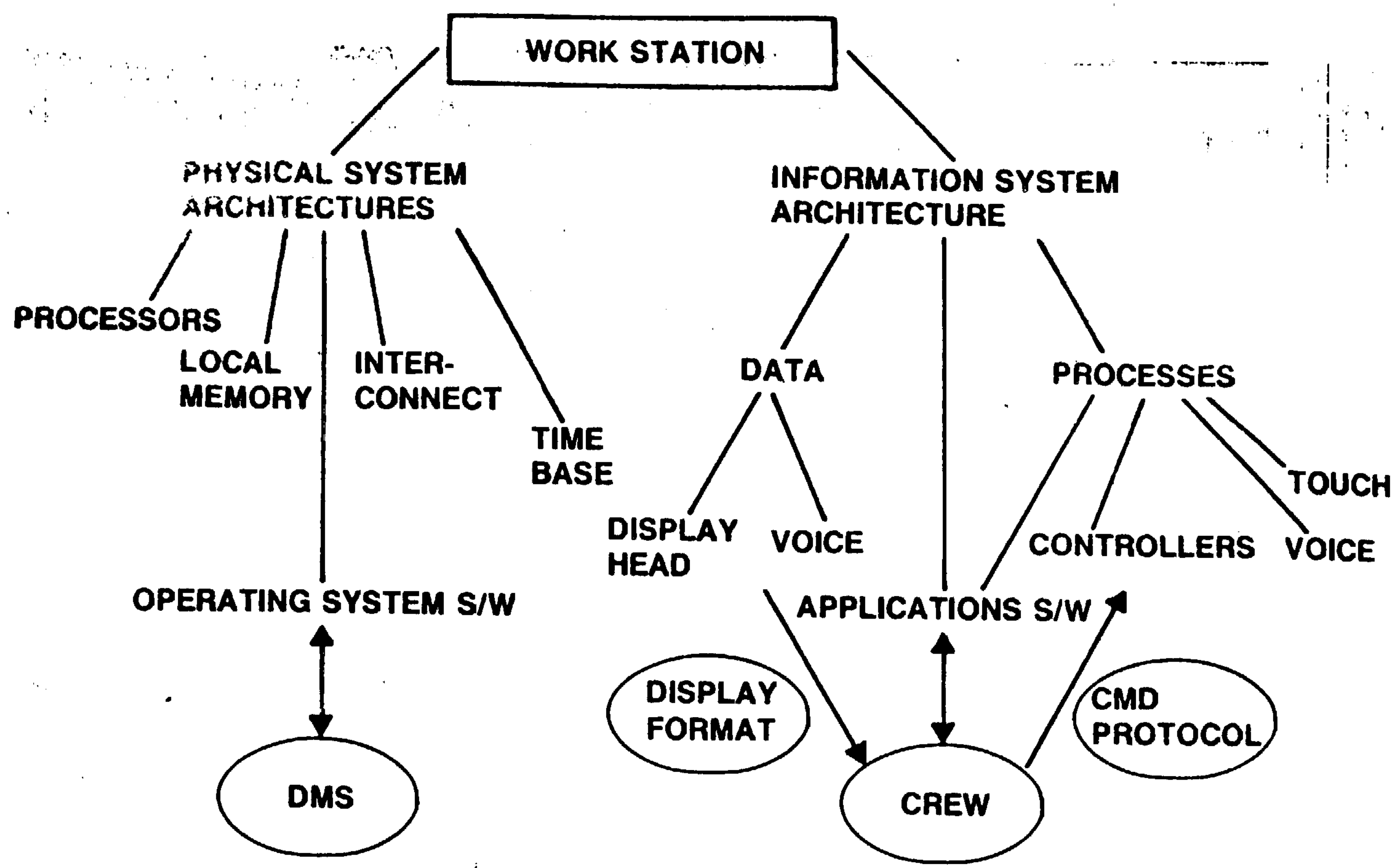
0190-007D

FUNCTIONAL WORK STATIONS

- P/L HANDLING MECHANISMS
  - RMS
  - HPA
  - CRANE/TRANSPORTER
  - CARGO BAY FIXTURES/LATCHES
  - BERTHING
- TELEOPERATOR
- FREEFLYER
  - OMV
  - MTV
  - MRWS (CLOSED/OPEN CABIN)
- EVA/MAINTENANCE/REPAIR (FIXED + IVA PORTABLE + EVA PORTABLE)
  - CONSTRUCTION
  - MAINT./REPAIR
  - EVA
- MISSION PLANNING/SCHEDULING/SUBSYSTEM MONITORING
- P/L/EXPERIMENTS/SPACE MANUFACTURING
- SUBSYSTEM MONITOR/CONTROL
- BIOMEDICAL

0190-008D

WORK STATION SYSTEM ARCHITECTURE CONCEPT



0190-009D

MAN MACHINE INTERFACE TECHNOLOGY

DISPLAYS:

- — COMPUTER COLOR GRAPHICS (ANALOG)
- — ALPHA/NUMERIC DATA (DIGITAL)
- — VOICE
- — CCTV
- — OTHER SENSOR (IE RADAR/INFRARED)
- — DIRECT VISION/STEREO OPTIC
- — HUD

PROCESSING:

- VIDEO IMAGING
- VOICE
- DATA
- PROCESS CONTROL
- PROCESS DISPLAY
- SPECIAL PURPOSE (ARRAY PROCESSOR)

CONTROLLERS:

- PROCESS CONTROLLERS (CARTESIAN HAND CONTROLLERS)
- TOUCH BEZEL
- VOICE
- INTEROCCULAR
- KEYBOARD/KEYPAD
- REPROGRAMABLE FUNCTION SWITCHES

0190-010D

WORK STATION SYSTEM STUDY FACTORS

DISPLAYS

- COLOR/MONOCHROME
- FLAT PANEL/CRT
- 2D/3D/STEREO OPTIC/HUD
- RESOLUTION/TONE/QUALITY
- SOFT/HARD COPY
- BANDWIDTH AND THRUPUT
- MAN MACHINE I/F
  - — INPUT/OUTPUT
  - — DISPLAY FORMATS
  - — CREATIVITY
  - — INTERNATIONAL CREW
- RELIABILITY
- GROWTH AND SCAR

"STRESS"

- CRITICAL FLICKER FREQ
- ANTI-ALIASING
- COLOR SATURATION

0190-011D

WORK STATION SYSTEM STUDY FACTORS (CONT)

INPUT/OUTPUT

- VISUAL
- TACTAL
- AUDITORY

- MICRO "g" ENVIRONMENT
- VIBRO ACOUSTIC NOISE
- INFRARED COMMUNICATIONS
- ANTHROPOMETRICS/ARCHITECTURE
- MAN-MACHINE I/F
  - GENERIC & KITABLE
  - MULTI-FUNCTIONAL
  - PRODUCTIVE
  - NATURAL
  - FRIENDLY
  - CREATIVE
  - LEVEL OF INTERACTION
  - DYNAMIC CALIBRATION
- RELIABILITY
- GROWTH & SCAR

0190-012D

WORK STATION SYSTEM STUDY FACTORS (CONT)

SOFTWARE

ONBOARD VALIDATION/VERIFICATION

- REALTIME OPERATING SYSTEM
  - AUTONOMOUS USERS NODE
  - SUPPORT CREATIVITY
  - MULTI-TASKING
  - CONFIGURATION CONTROL (LOCAL)
  - TRANSPARENT I/O OPERATIONS
  - SYNCHRONIZATION
  - FAULT ISOLATION (LOCAL)
- APPLICATIONS SOFTWARE SEMANTIC
  - IMBEDDED SYNTAX & STRUCTURE
  - DISPLAY FORMATS
  - CREATIVE AIDS
  - CMD/DATA PROCESSING
  - SIMULATION/TRAINING/VALIDATION
  - PLANNING/SCHEDULING/MANIFESTING
  - IMAGE PROCESSING
  - VOICE PROCESSING

0190-013D

RECOMMENDATIONS & CONCLUSIONS

- DEVELOP PROXIMITY OPERATIONS DESIGN REFERENCE MISSIONS FOR C & D MISSION ANALYSIS
- DEVELOP C & D TEST BED & DYNAMIC ENVIRONMENT SIMULATOR TO EVALUATE DESIGN CONCEPTS
- DEFINE TRADE STUDIES & EVALUATE RISK
- DEVELOP GENERIC DESIGN CONCEPTS TO EVALUATE PRODUCTIVITY & DEFINE TECHNOLOGY RESOURCE REQUIREMENTS
  - COMPUTER GRAPHICS
  - DISPLAY HEADS
  - VOICE
  - TOUCH BEZEL
  - SMART SOFTWARE

• GENERIC TECHNOLOGY
• VALIDATED CONCEPT
• CLEARLY DEFINED RISK

- EVALUATE DESIGN CONCEPTS BY SIMULATION TEST
- DEFINE WORK STATION C & D REQUIREMENTS FOR PHASE C/D DESIGN DEVELOPMENT

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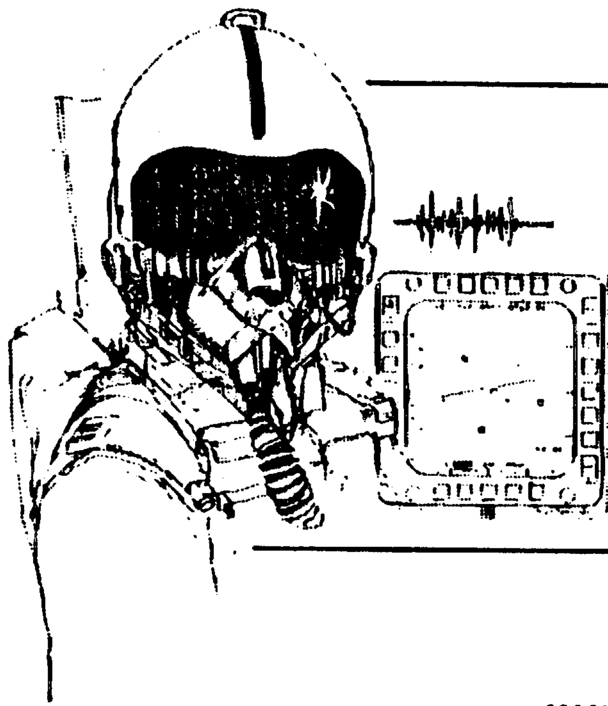

APPLICATIONS OF VOICE INTERACTIVE SYSTEMS FOR THE FUTURE

CAROLYN A. MOORE
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VICE PRESIDENT OF MARKETING AND
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MCDONNELL DOUGLAS ELECTRONICS COMPANY
ST. CHARLES, MISSOURI

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- **GENERAL VRS APPLICATIONS**
- SINGLE-SEAT FIGHTER VRS CONSIDERATIONS
  - REQUIREMENTS
  - CANDIDATE FUNCTIONS
  - PILOT-VEHICLE INTERFACE DESIGN
  - ARCHITECTURE AND SUBSYSTEM DEFINITION/INTEGRATION
- EXAMPLE IMPLEMENTATIONS
  - AFTI/F-16 PHASE I
  - AFTI/F-16 PHASE II
  - SPACEBORNE APPLICATIONS
- CONCLUSIONS

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THE SUCCESS OF VOICE RECOGNITION AND SYNTHESIS (VRS) IN THE SPECIFIC ENVIRONMENT DEPENDS UPON ANSWERING THESE KEY QUESTIONS:

(1) IS VRS A VIABLE ALTERNATIVE TO TRADITIONAL MANUAL CONTROL AND INFORMATION FEEDBACK METHODS?

(2) CAN VRS BE MADE RELIABLE AND INTELLIGIBLE IN THE SPECIFIC ENVIRONMENT?

(3) WHICH FUNCTIONS SHOULD BE VOICE CONTROLLED AND WHICH OFFER THE HIGHEST PAYOFFS?

(4) WHICH WORDS/PHRASES SHOULD BE SYNTHESIZED TO ACHIEVE APPROPRIATE INFORMATION TRANSFER?

(5) HOW SHOULD POTENTIAL VRS FUNCTIONS BE CONTROLLED?

(6) WHAT ARE THE IMPLICATIONS OF UTILIZING SYNTHESIS AND RECOGNITION TOGETHER?

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APPLICATION EXAMPLES

- DATA BASE MANAGEMENT SYSTEMS
  - TAC FIRE
  - PHOTOGRAPHIC INTERPRETER
  - INTEGRATED INFORMATION DISPLAY
  - ELECTRONIC PARTS TRACEABILITY
  - MAINTENANCE AIDING

- COMMAND AND CONTROL OF WEAPON SYSTEMS
  - AFTI-F-16 FLIGHT TEST
  - VOICE RECOGNITION AND SYNTHESIS (VRAS)

- TRAINING SYSTEMS
  - PRECISION APPROACH RADAR TRAINING SYSTEM AND AIR CONTROLLER EXERCISE

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TYPICAL VOICE INTERACTIVE SYSTEM CHARACTERISTICS BY FUNCTIONAL APPLICATION

	DATA BASE MANAGEMENT SYSTEM	COMMAND AND CONTROL OF WEAPONS SYSTEMS	TRAINING
VOCABULARY SIZE IN WORDS	LARGE 1000-5000	SMALL LESS THAN 100	MEDIUM 50-500
RECOGNIZER TYPE	CONNECTED UTTERANCE	DISCRETE WITH CONNECTED DIGIT CAPABILITY	DISCRETE WITH CONNECTED DIGIT CAPABILITY
SPEAKER DEPENDENCE	SPEAKER INDEPENDENT	SPEAKER DEPENDENT	SPEAKER DEPENDENT
ENROLLMENT TIME	NOT APPLICABLE	VARIES	LESS THAN 5% OF TOTAL TRAINING TIME
TYPICAL NOISE	QUIET	MODERATE TO HIGH	QUIET
TYPICAL OPERATOR STRESS	NONE TO MODERATE	MODERATE TO HIGH	MODERATE TO HIGH

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from National Research Council,
Committee on Computerized Speech Recognition
<u>Automatic Speech Recognition in Severe Environments</u>
Washington, D.C.: National Academy Press, 1984

TYPICAL VOICE INTERACTIVE SYSTEM CHARACTERISTICS BY FUNCTIONAL APPLICATION (CONTINUED)

	DATA BASE MANAGEMENT SYSTEM	COMMAND AND CONTROL OF WEAPONS SYSTEMS	TRAINING
OPERATIONAL REQUIREMENTS*	LESS THAN 5% ERROR	LESS THAN 1% ERROR	LESS THAN 3% ERROR
VIS RESPONSE TIME	LESS THAN 5 SECONDS	LESS THAN 1 SECOND	LESS THAN 2 SECONDS
SYSTEM INTEGRATION REQUIREMENTS	MODERATE	CRITICAL	IMPORTANT
PHYSICAL CON-STRAINTS (SIZE, WEIGHT, POWER, COOLING)	MINIMAL	SEVERE	MINIMAL

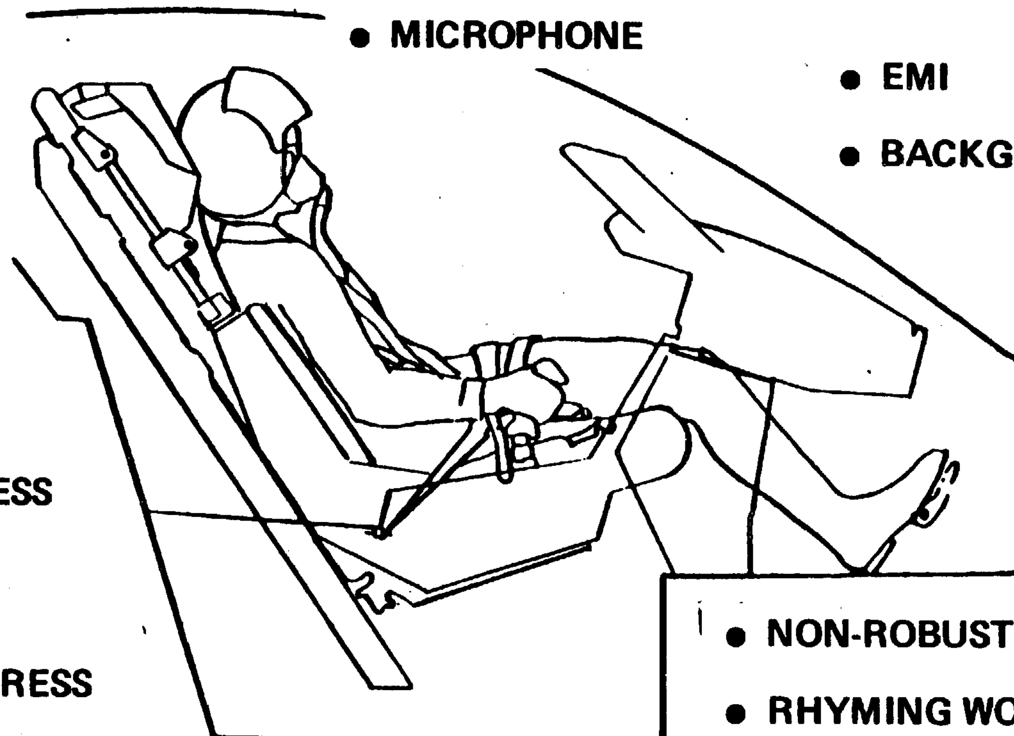
*THERE ARE CERTAIN SAFETY AND SURVIVABILITY CONDITIONS THAT MANDATE MINIMAL ERROR TOLERANCE FOR PORTIONS OF THE VOCABULARY

from National Research Council,
Committee on Computerized Speech Recognition
<u>Automatic Speech Recognition in Severe Environments</u>
Washington, D.C.: National Academy Press, 1984

AIRCRAFT ENVIRONMENT CONSIDERATIONS FOR VOICE RECOGNITION

✔ NUMEROUS POTENTIAL ERROR SOURCES EXIST

- OXYGEN MASK
- MICROPHONE
- EMI
- BACKGROUND NOISE



- PHYSICAL STRESS
  - ALTITUDE
  - G FORCES
- EMOTIONAL STRESS
  - COMBAT SITUATION

- NON-ROBUST WORDS
- RHYMING WORDS
- SPEAKER DEPENDENT SPEECH PECULIARITIES

- AFFECT VOICE RECOGNIZER'S ABILITY TO INTERPRET SPOKEN WORDS

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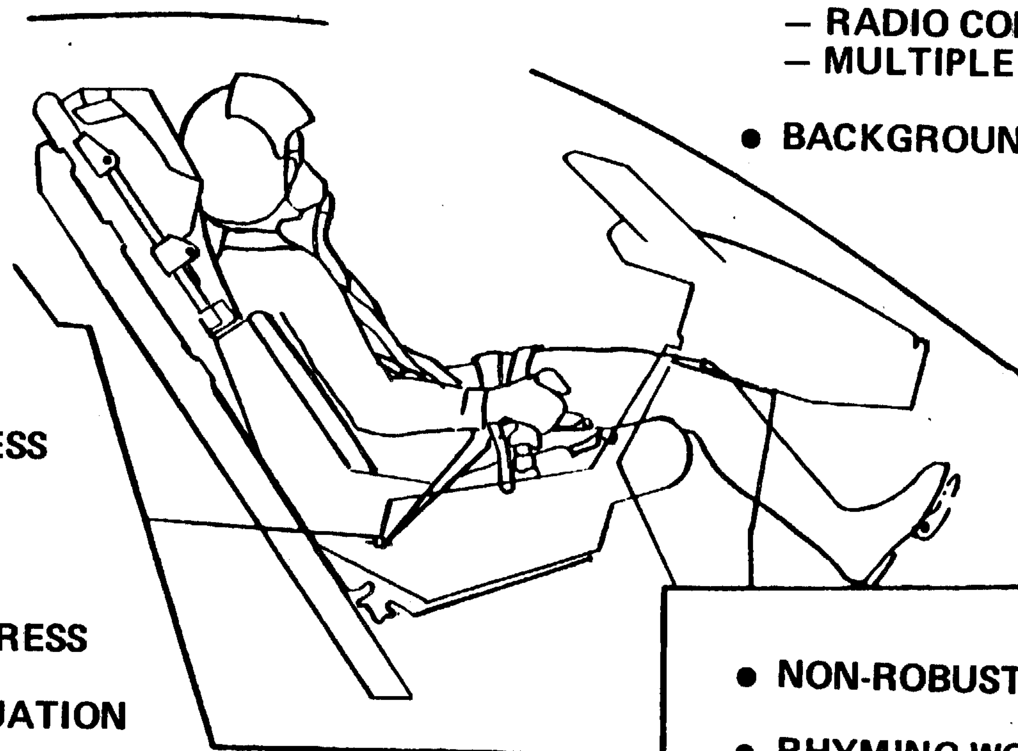
AIRCRAFT ENVIRONMENT CONSIDERATIONS FOR SPEECH GENERATION

NUMEROUS POTENTIAL ERROR SOURCES EXIST

- MULTIPLE VOICES IN COCKPIT
  - RADIO COMMUNICATIONS
  - MULTIPLE CREW MEMBERS
- BACKGROUND NOISE



- PHYSICAL STRESS
  - ALTITUDE
  - G FORCE
- EMOTIONAL STRESS
  - COMBAT SITUATION

- NON-ROBUST WORDS
- RHYMING WORDS
- MULTI-NATIONAL USERS

- AFFECT PILOT'S ABILITY TO HEAR AND UNDERSTAND

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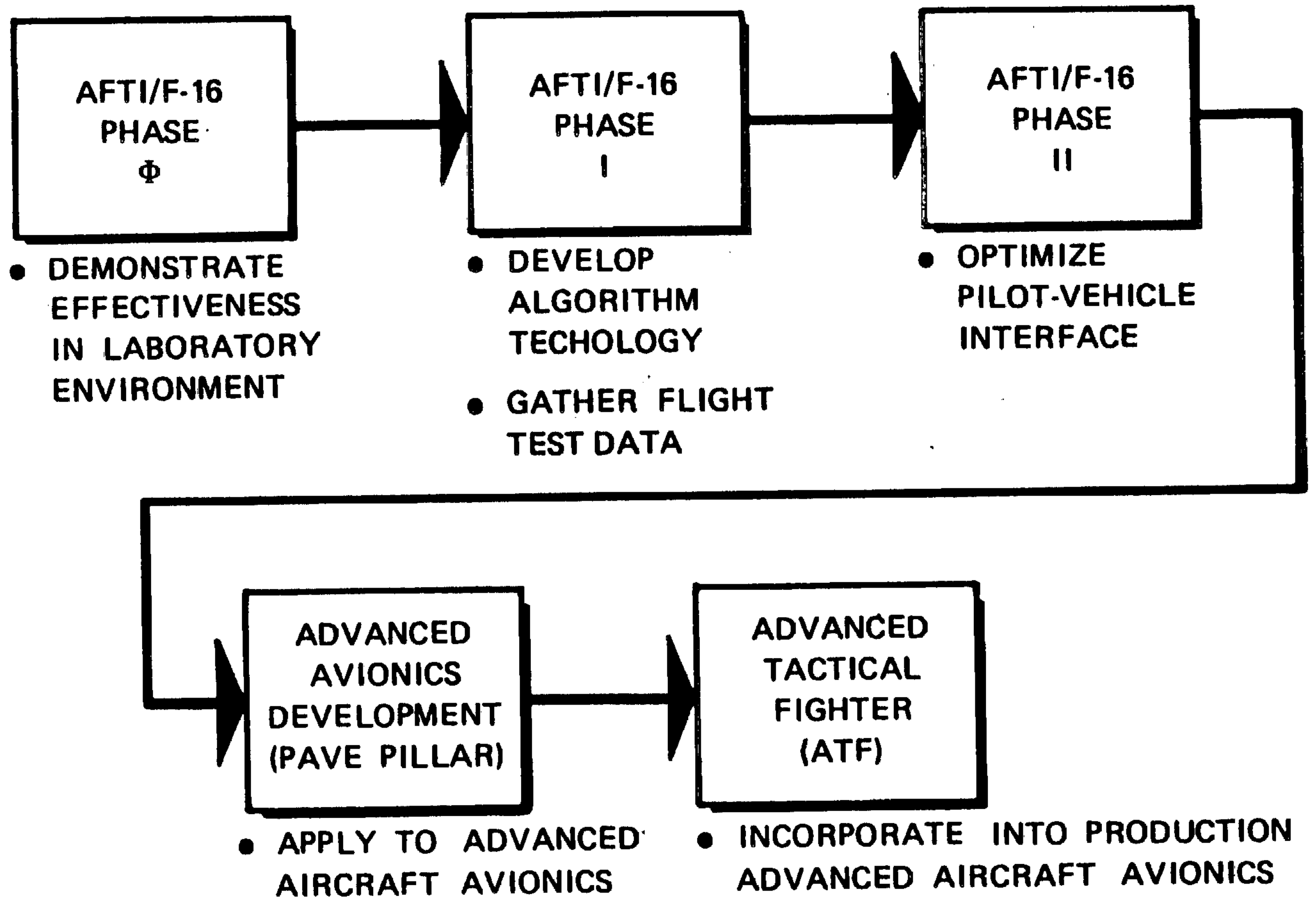
PROGRESSIVELY IMPLEMENT INTERACTIVE VOICE INTO ADVANCED AVIONIC SYSTEMS



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- GENERAL VRS APPLICATIONS
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- CONCLUSIONS

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PILOT-VEHICLE INTERACTION PHILOSOPHY

- NUMBER OF SYSTEM INPUTS AND OUTPUTS MUST BE KEPT TO A MINIMUM TO ALLOW SIMPLE, EFFICIENT OPERATION
- DO NOT DISTRIBUTE BASIC SYSTEM FUNCTIONS OVER VARIOUS SUBSYSTEMS (KEEP HIGH COHESION WITHIN EACH SUBSYSTEM)
- WEIGHT SYSTEM MECHANIZATION (TASK-TAILORING) SO TIME CRITICAL AND HIGH PAYOFF FUNCTIONS TAKE THE SHORTEST AMOUNT OF TIME TO IMPLEMENT
- THE SYSTEM AND PVI SHOULD HAVE CONSISTENT MECHANIZATION
- APPLY ARTIFICIAL INTELLIGENCE TECHNIQUES TO THE SYSTEM DESIGN AND PVI OPERATION (DEFAULT STRUCTURES)
- THE PILOT—VEHICLE INTERFACE SHOULD EMULATE HUMAN INTERACTION

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CANDIDATE FUNCTIONS FOR ADVANCED AIRCRAFT

- INFORMATION ACQUISITION
  - SENSORS
  - COMM/NAV
  - DATABASES/DIGITAL MAP
- INFORMATION FUSION
- MISSION PHASE SELECTION/CONTROL
- WEAPONS/COUNTERMEASURES SELECTION/CONTROL
- AIRCRAFT SUBSYSTEM CONTROL
- MISSION/AIRCRAFT MANAGEMENT

VRS FUNCTIONS SHOULD BE HIGHLY RELIABLE, DISCRETE QUANTITIES, QUALITIES, OR CONCEPTS

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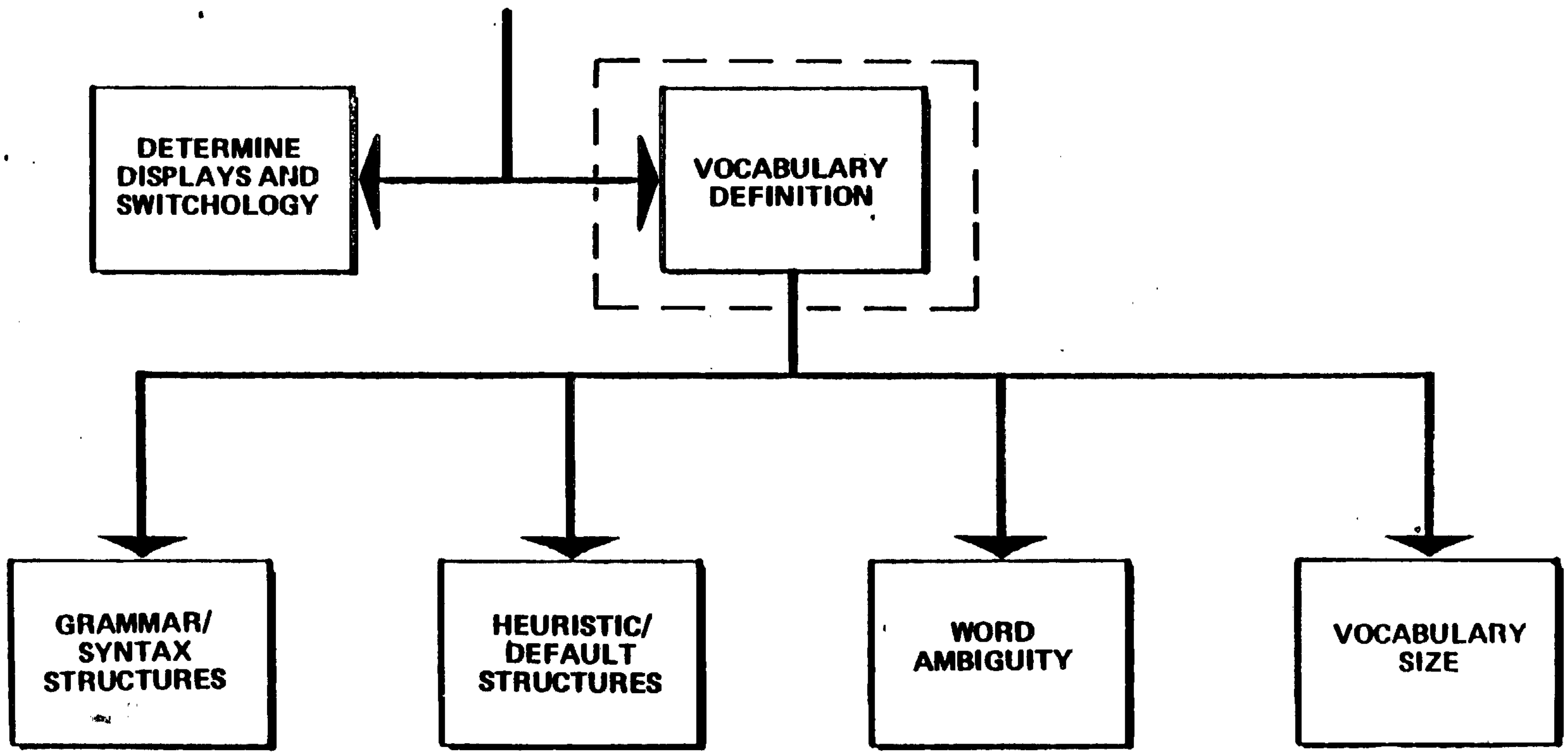
PILOT-VEHICLE INTERFACE DESIGN



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USE NATURAL GRAMMAR TO ACHIEVE NATURAL OPERATION

VOICE RECOGNITION

A NATURAL GRAMMAR STRUCTURE IS EASY TO USE SINCE:

- PILOT DOES NOT HAVE TO LEARN AN UNNATURAL LANGUAGE STRUCTURE
- PILOT DOES NOT HAVE TO REMEMBER A SET OF UNRELATED COMMANDS
- LANGUAGE STRUCTURE CAN PROMPT PILOT ON WHAT TO SAY NEXT

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SYNTAX STRUCTURES AND NODE DEFINITION

VOICE RECOGNITION

- A SYNTAX STRUCTURE SHOWS THE INTERRELATIONSHIPS BETWEEN ALL SYNTAX NODES
- A SYNTAX NODE IS THE PARTITIONING OF ALLOWABLE VOCABULARY WORDS BY GRAMMATICAL ELEMENT AND AIRCRAFT AVIONIC STATE
  - E.G., WHAT IS THE LIST OF ACCEPTABLE DISPLAYS IN AIR-TO-GROUND WEAPON DELIVERY MODE?
- A SYNTAX STRUCTURE/NODE IS USED TO:
  - DECREASE RECOGNITION RESPONSE TIME
  - INCREASE RECOGNITION ACCURACY

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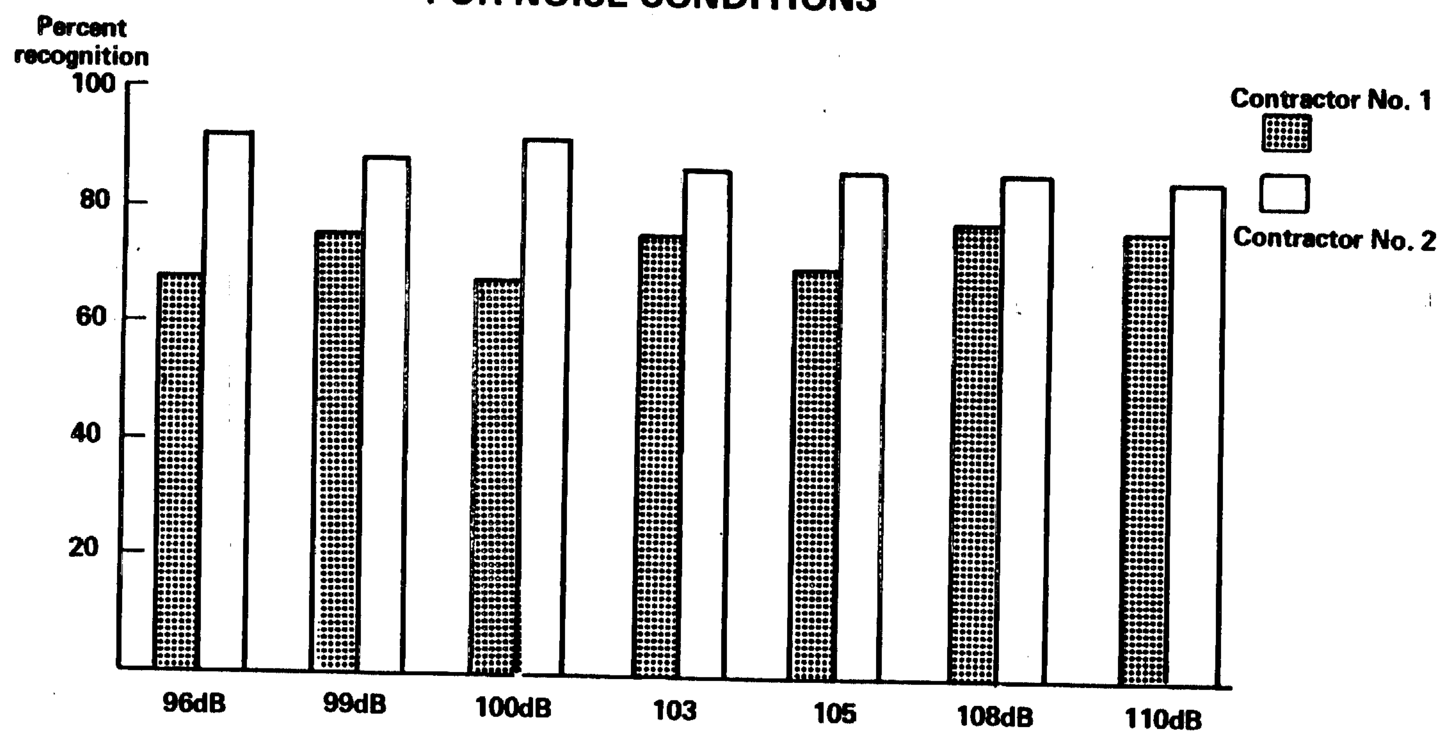
VOICE COMMAND SYSTEM

INFLIGHT RECOGNITION ACCURACY FOR NOISE CONDITIONS



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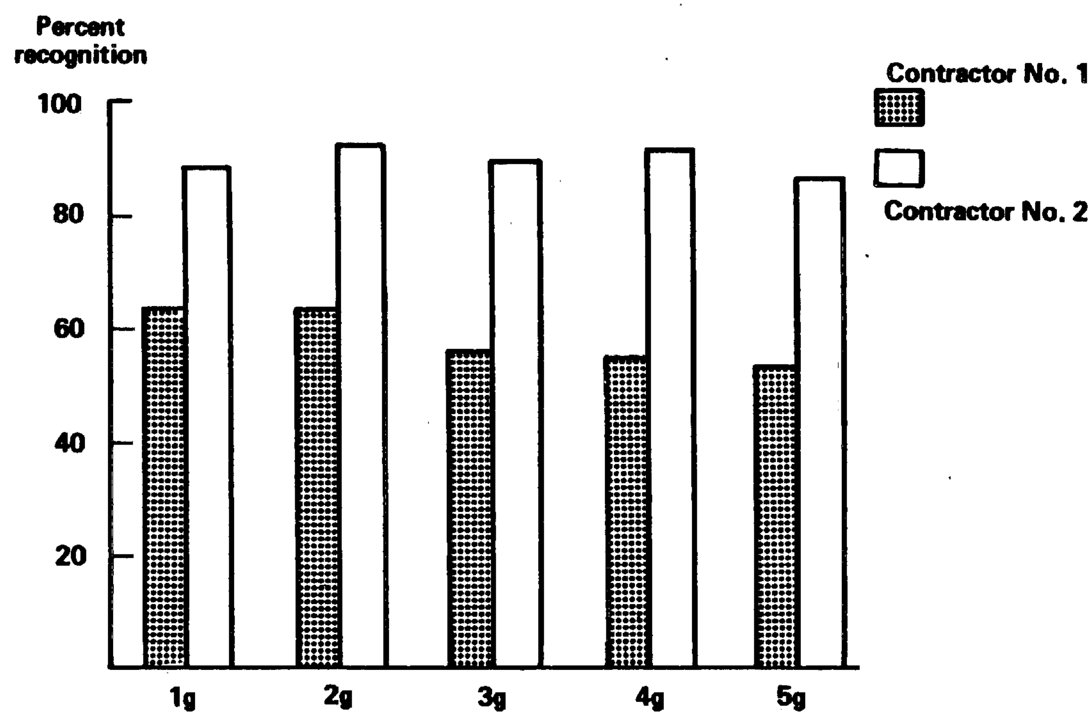
VOICE COMMAND SYSTEM

INFLIGHT RECOGNITION ACCURACY FOR LOAD FACTOR CONDITIONS



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SPEECH GENERATION GRAMMAR STRUCTURE

BASIC VOCABULARY COMPONENTS

EXAMPLE:

- WORDS
- PHRASES
- LISTS
- TONES

BASIC RESPONSE TYPES

EXAMPLE:

- WARNING
- CAUTION
- STATUS
- RESPONSE TO A QUERY
- "SAY AGAIN" REQUEST

DERIVE GRAMMAR STRUCTURE

REFINE GRAMMAR STRUCTURE

SPEECH GENERATION

- **PRIORITIZE VOCABULARY COMPONENTS**

  EXAMPLE:

  1. **WARNING**
  2. **CAUTION**
  3. **STATUS**
  4. **RESPONSE**
  5. **"SAY AGAIN" REQUEST**

- **ESTABLISH SYNTHESIS CONVENTIONS/RULES**

  – NORMAL INPUT/OUTPUT SOUND CONVENTIONS
  – VARIABLE SOUND OUTPUT CONVENTIONS
  – INTERRUPT CONVENTIONS

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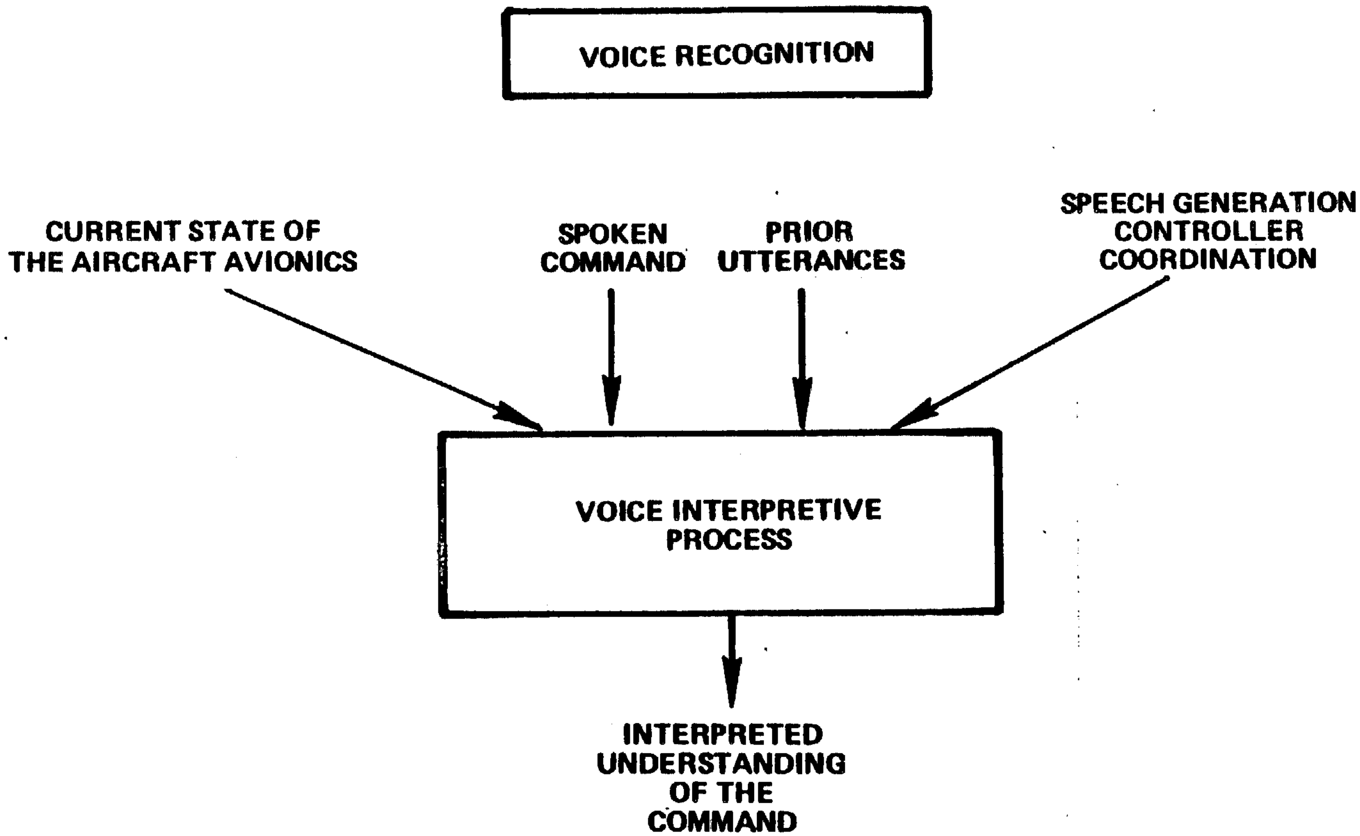
ARTIFICIAL INTELLIGENCE APPLICATIONS



- DEGREE OF COMPLEXITY IS DIRECTLY RELATED TO TO THE SOPHISTICATION OF THE VOICE INTERPRETER AND THE AIRCRAFT AVIONICS

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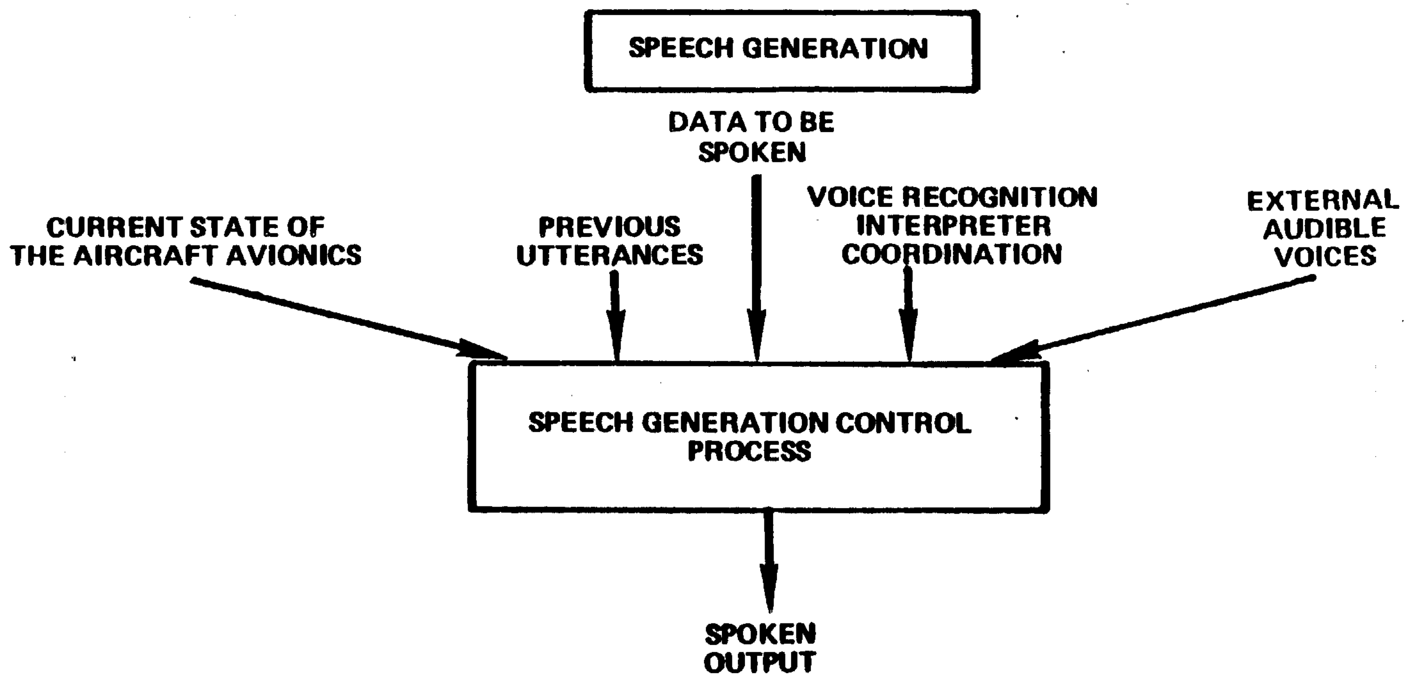
ARTIFICIAL INTELLIGENCE APPLICATIONS



- DEGREE OF COMPLEXITY IS DIRECTLY RELATED TO THE SOPHISTICATION OF THE SPEECH GENERATION CONTROLLER AND THE AIRCRAFT AVIONICS

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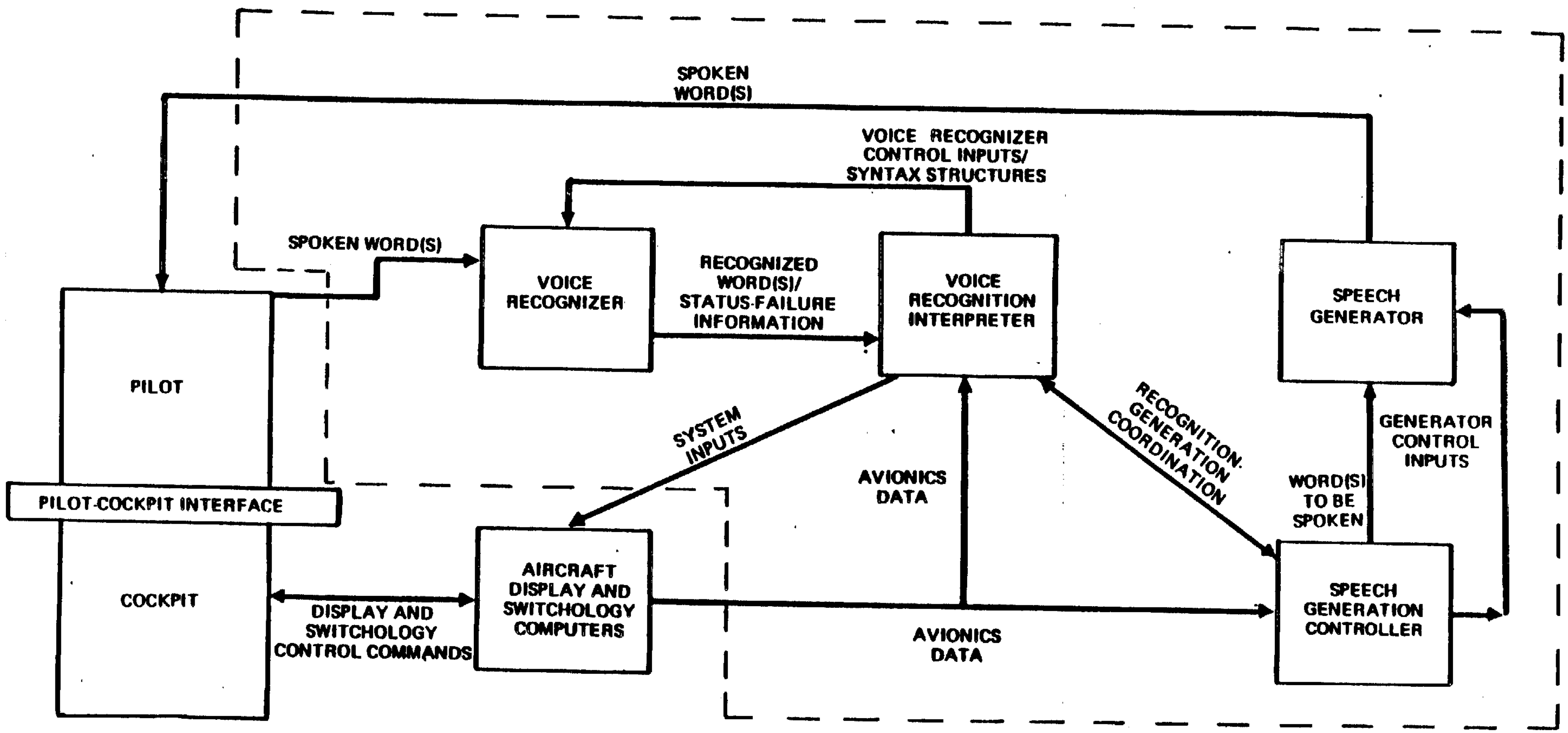
INTEGRATION OF VOICE RECOGNITION AND SPEECH GENERATION INTO AN ADVANCED AIRCRAFT



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- CONCLUSIONS

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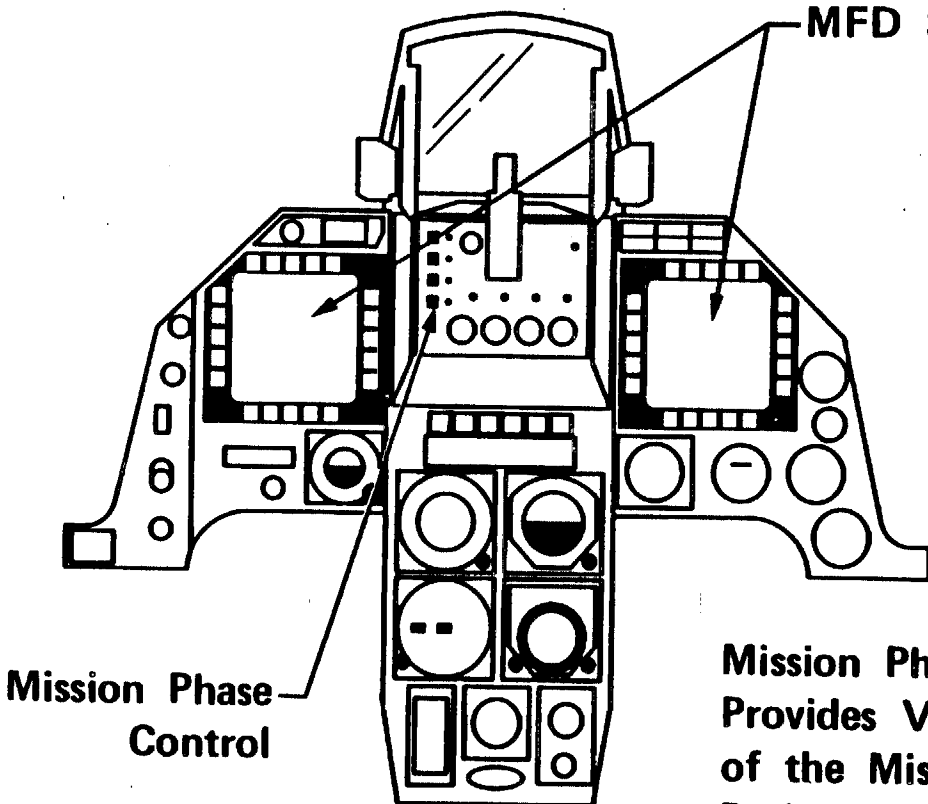
7-111 C-6

AFTI/F-16 Cockpit Display



MFD Switch Control—Provides Complete Voice Command Control of the MFDs by Assigning a Voice Command Word for Each Switch on the MFD. This Allows Control of Stores Management, Flight Control Functions and Video Selection

Mission Phase Control—Provides Voice Command Control of the Mission Phase Pushbuttons

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AFTI/F-16 Phase I Voice Command Vocabulary

Commands	Function
Air-To-Air Strafe Missile Air-To-Surface Navigation (NAV)	Mission Phase Selection
Left Right	Specific MFD Selection
S-M-S F-C-S V-C-S Radar RAW	MFD Page Selection
Fault Data Test Pre-Set Authority Inventory Load	MFD Page Option Control
Enter Back Clear Space Exit	Data Entry Option Control
One Two Three Four Five Six Seven Eight Nine Zero	MFD Button/ Numeric Data Entry Selection

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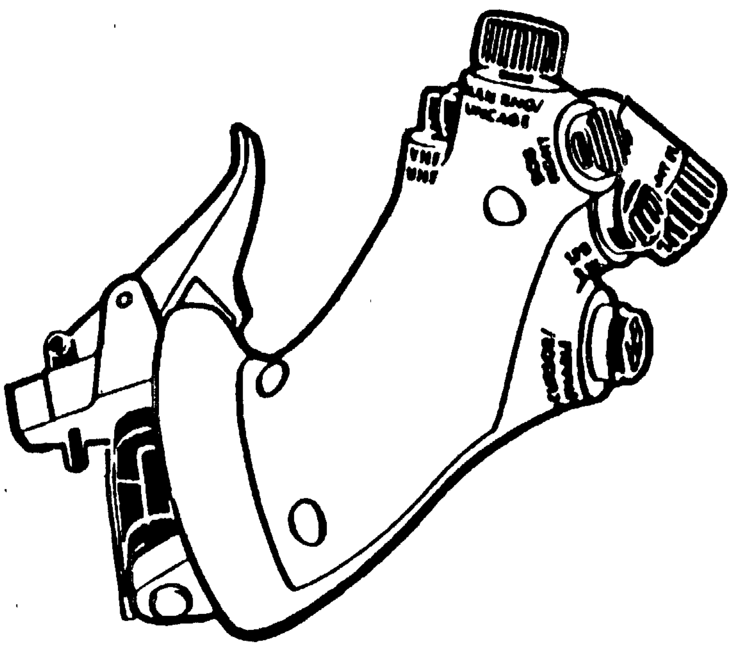
VOICE ENABLE PREVENTS SYSTEM REACTION TO EXTRANEOUS INPUTS



- "HOT" UHF MIKE ENABLES USAGE OF UHF SWITCH ON THROTTLE
- DEPRESSING THE SWITCH ENABLES THE VCS TO LISTEN AND RESPOND TO AUDIO INPUTS FROM THE OXYGEN MASK MICROPHONE
- RELEASING THE SWITCH DESABLES THE VCS FROM LISTENING TO AUDIO INPUTS
- DEPRESSING THE SWITCH DISPLAYS THE VOICE ENABLE CUE (A SMALL SQUARE) ON THE HUD

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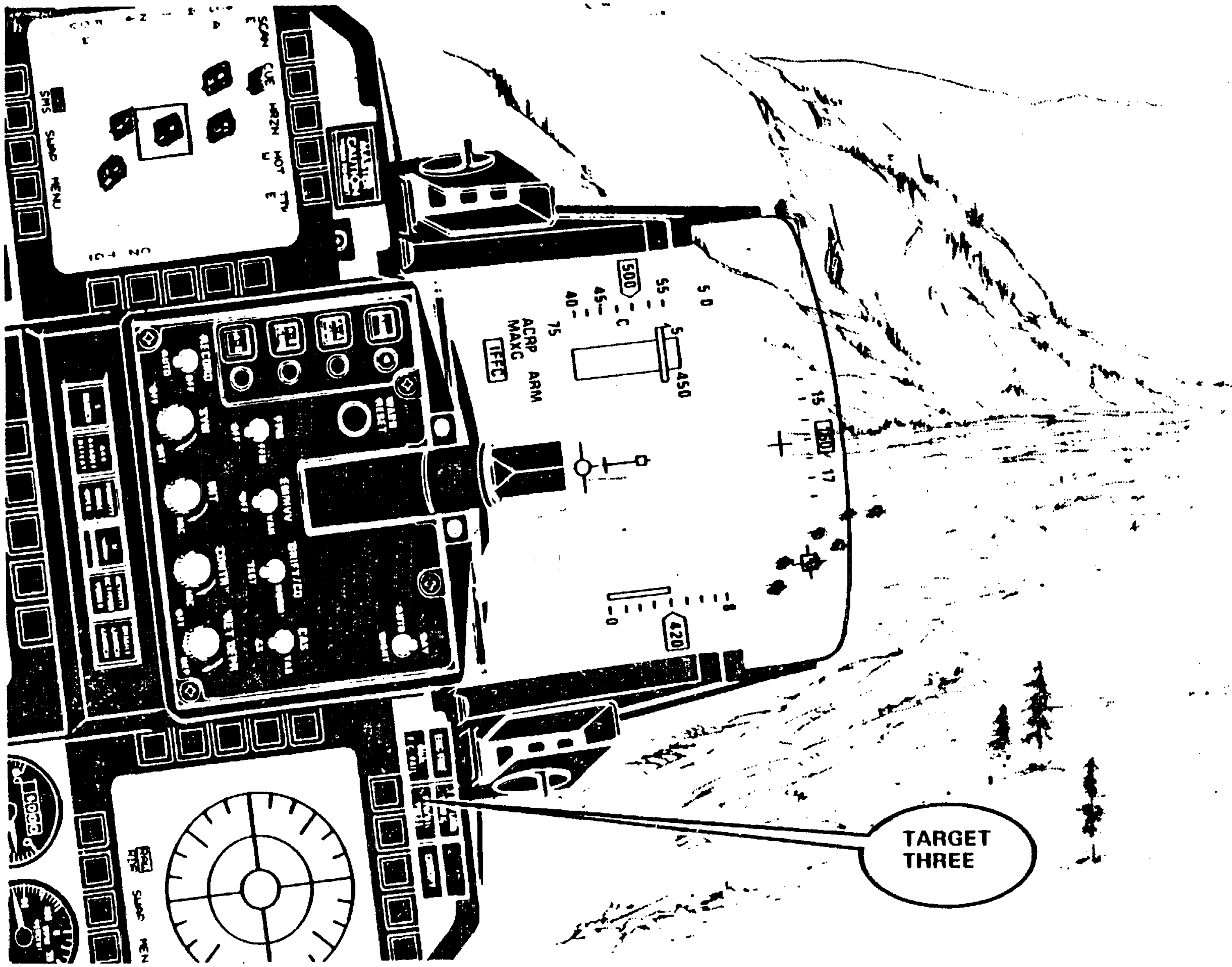
HOW DO WE MECHANIZE THE INTERACTIVE VOICE COCKPIT TO IMPROVE PILOT-AIRCRAFT OPERATION?



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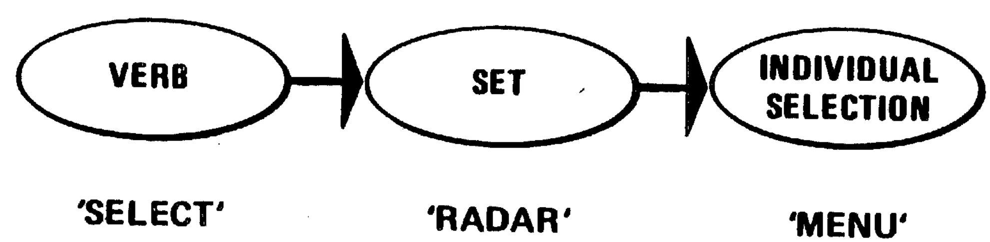
MOST VOICE FUNCTIONS CAN BE PERFORMED BY SELECTING/CANCELING OBJECTS

IN ORDER TO SELECT/CANCEL AN INDIVIDUAL OBJECT FROM A SET, WE MUST:

- IDENTIFY VERB (e.g., Select or Cancel)
- IDENTIFY SET TO BE ACTED UPON
- IDENTIFY INDIVIDUAL SELECTION



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DEFAULT VALUES

IN ORDER TO ADD INTELLIGENCE TO THE VCS AND SPEED UP SYSTEM OPERATION, DEFAULT VALUES ARE ASSIGNED TO PARTICULAR GRAMMATICAL ELEMENTS IN VOICE COMMAND SENTENCES

- VCS CAN MAKE LOGICAL ASSUMPTIONS OR ACQUIRE INFORMATION FROM OTHER SYSTEMS IN ORDER TO PROCESS THE COMMAND

- DEFAULT VALUES SHOULD ALWAYS BE NON-CRITICAL, NON-DESTRUCTIVE, UNAMBIGUOUS COMMANDS THAT WHEN ASSUMED WILL NEVER CAUSE AN ADVERSE SYSTEM RESPONSE

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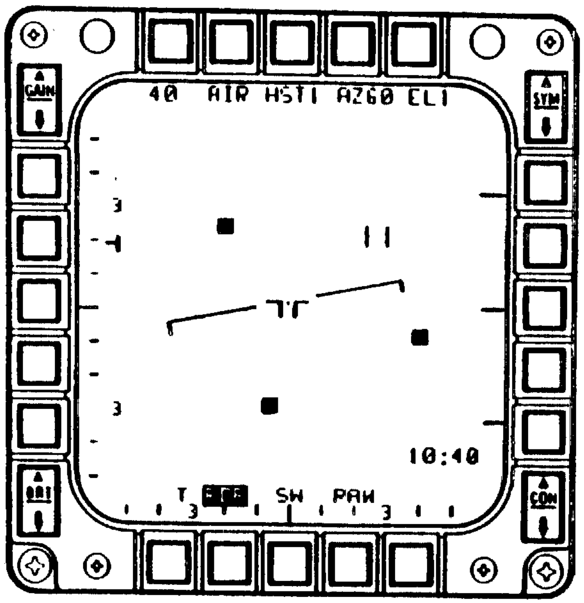
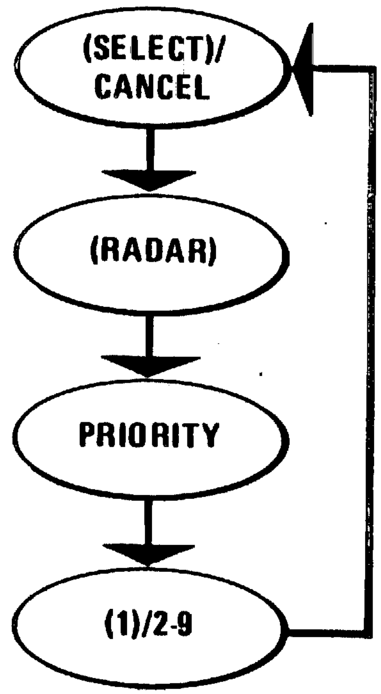
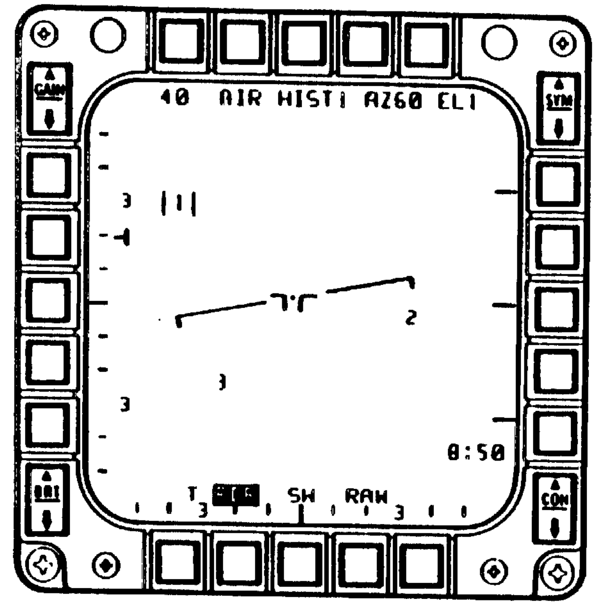
A/A RADAR LOCK VOICE COMMANDS

A/A RADAR PAGE





A/A RADAR
WITH TARGET 1 SELECTED



( ) = DEFAULT VALUE: OPTIONAL WORDS THAT CAN BE SPOKEN OR IMPLIED

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- CONCLUSIONS

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CONCLUSION

- VOICE COMMAND IS A VIABLE COCKPIT CONTROL TECHNOLOGY
  - VOICE RECOGNITION ALGORITHMS MUST CONTINUE TO IMPROVE
    - CONTINUOUS SPEECH CAPABILITY
    - SPEAKER INDEPENDENCE
    - OPERATE WELL IN SEVERE AIRBORNE ENVIRONMENT
  - VOICE INTEGRATION TECHNOLOGIES/METHODOLOGIES MUST BE ABLE TO OPTIMIZE VOICE COMMAND SYSTEM—PILOT INTERACTION
- SPEECH SYNTHESIS IS A VIABLE COCKPIT INFORMATION FEEDBACK TECHNOLOGY
  - COMPLEMENTS INFORMATION TRANSFER TO THE PILOT WHEN INTEGRATED PROPERLY INTO THE AVIONICS SUITE
- CURRENT AVIONIC SYSTEMS NEED TO TOTALLY INTEGRATE VOICE RECOGNITION AND VOICE SYNTHESIS TECHNOLOGIES

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SPACEBORN APPLICATIONS

- ALLOW VISUAL FOCUS TO REMAIN ON PRIMARY OBJECT OF ATTENTION DURING CRITICAL PERIODS RATHER THAN ON OTHER CONTROLS AND DISPLAYS (REDUCE WORKLOAD PEAKS)
- PROVIDE ALARMS THAT CONVEY TIME CRITICAL DISCRETE DATA E.G. SPACESTATION: "MODULE 3 AIR LEAK, COMPARTMENT SEALING REQUIRED"
- PROVIDE EASIER USE/ACCESS OR PROVIDE CAPABILITIES CURRENTLY NON-EXISTANT FOR SPACE SUIT CONTROL/DISPLAY FUNCTIONS

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THE HUMAN ROLE IN SPACE

THE HUMAN ROLE IN SPACE (THURIS) STUDY

VGU096

Objectives

- To Investigate Role and Degree of Direct Involvement of Humans in Future Space Missions
- To Establish Criteria for Allocation of Functional Activities Between Humans and Machines
- To Provide Insight Into Technology Requirements, Economics, and Benefits of the Human Presence in Space

Results

- Developed Methodology for Space Activity Allocation Based on Criteria of Performance, Cost, and Technological Readiness

GENERIC ACTIVITIES IN PROXIMITY OPERATIONS

1. ACTIVATE SYSTEMS
3. ALLOCATE RESOURCES
5. COMMUNICATE INFORMATION
8. CONFIRM OPERATIONS
11. CORRELATE DATA
12. DEACTIVATE SYSTEMS
13. DECODE DATA
15. DEPLOY APPENDAGES
17. DISPLAY DATA
20. IMPLEMENT PROCEDURES
22. INSPECT/OBSERVE
27. MAKE DECISIONS
28. PURSUIT TRACKING

572–85

PRESENTATION OVERVIEW

- GENERAL BENEFITS OF MANNED INVOLVEMENT
- EXAMPLE OF THURIS ANALYSIS APPROACH:
  GENERIC ACTIVITY FROM TYPICAL PROJECT
- APPLICATION TO OVERALL PROJECT
- CONCLUDING REMARKS

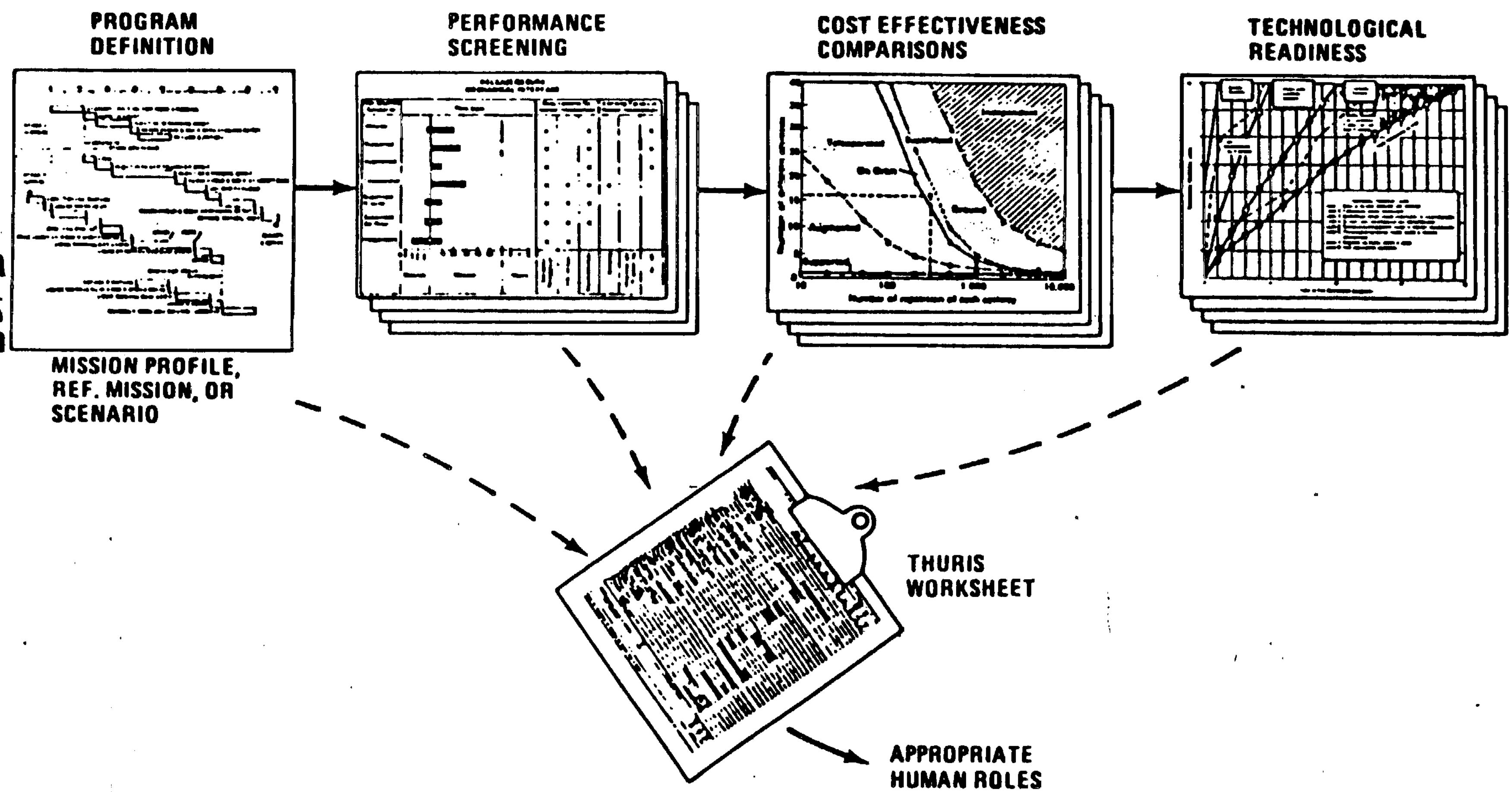
PROXIMITY OPERATIONS GENERIC ACTIVITIES

NO.	GENERIC SPACE ACTIVITY	KEY HUMAN CAPABILITIES	BENEFITS OF MAN'S ONBOARD PARTICIPATION		
			EQUIPMENT CAN BE ELIMINATED	PERFORMANCE OF ACTIVITY IS IMPROVED	PROBABILITY OF MISSION SUCCESS INCREASED
22.	INSPECT/ OBSERVE	• VISUAL ACTIVITY • BRIGHTNESS DISCRIMINATION • DEPTH PERCEPTION • VISUAL ACCOMMODATION	YES	YES	YES

OVERALL BENEFIT FROM MAN'S ONBOARD PARTICIPATION	RATIONALE
HIGHLY BENEFICIAL	MAN'S SELECTIVE OBSERVATIONS SUPERIOR TO AUTOMATED MONITORING

189–85

THE HUMAN ROLE IN SPACE (THURIS) ANALYSIS



7-127

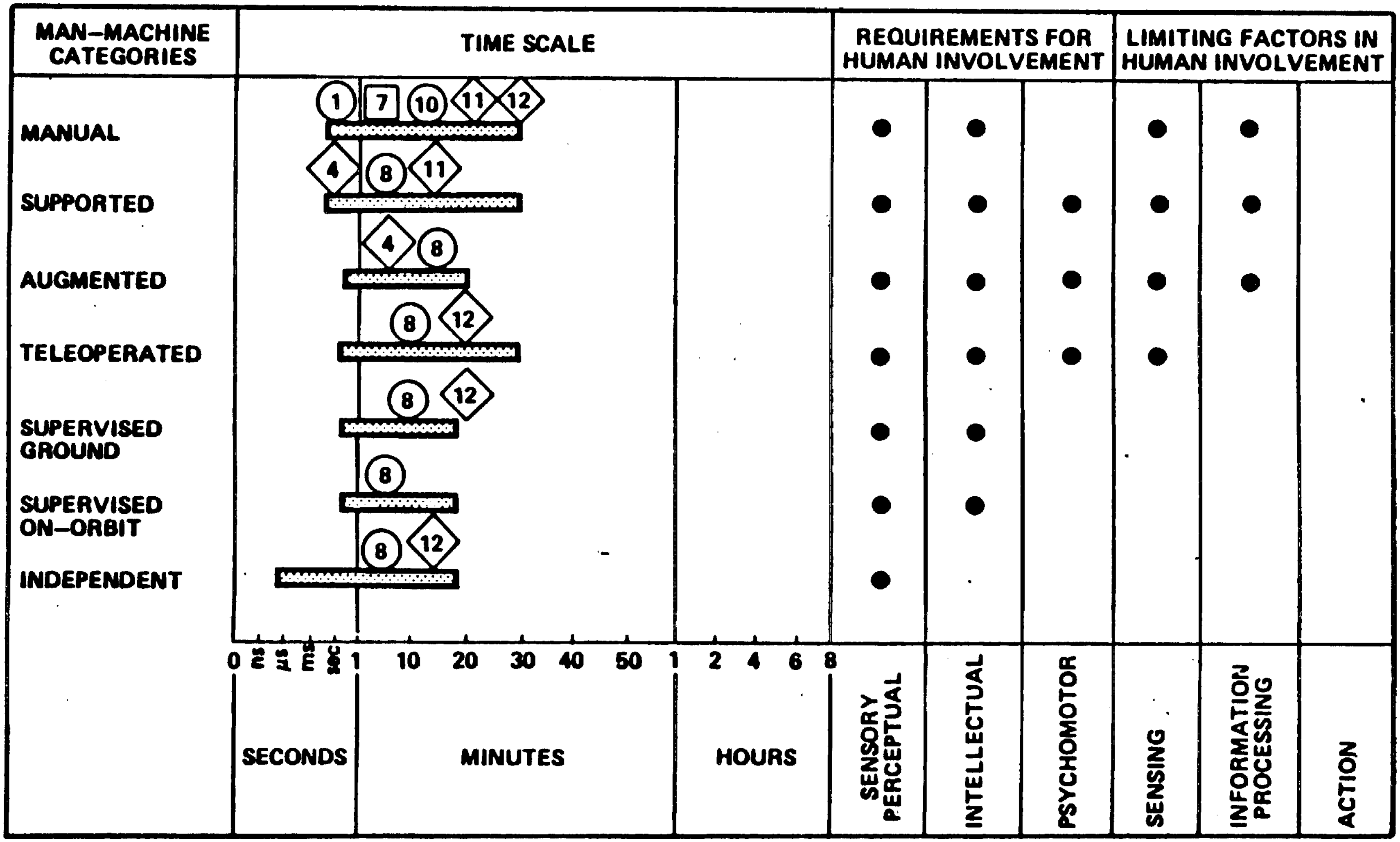
CATEGORIES OF MAN-MACHINE INTERACTION

VGL403

MANUAL	Unaided IVA/EVA, with Simple (Unpowered) Hand Tools
SUPPORTED	Requires Use of Supporting Machinery or Facilities to Accomplish Assigned Tasks (e.g., Manned Maneuvering Units and Foot Restraint Devices)
AUGMENTED	Amplification of Human Sensory or Motor Capabilities (Powered Tools, Exo-Skeletons, Microscopes, etc.)
TELEOPERATED	Use of Remotely Controlled Sensors and Actuators Allowing the Human Presence to Be Removed From the Work Site (Remote Manipulator Systems, Teleoperators, Telefactors)
SUPERVISED	Replacement of Direct Manual Control of System Operation with Computer-Directed Functions Although Maintaining Humans in Supervisory Control
INDEPENDENT	Basically Self-Actuating, Self-Healing, Independent Operations Minimizing Requirement for Direct Human Intervention (Dependent on Automation and Artificial Intelligence)

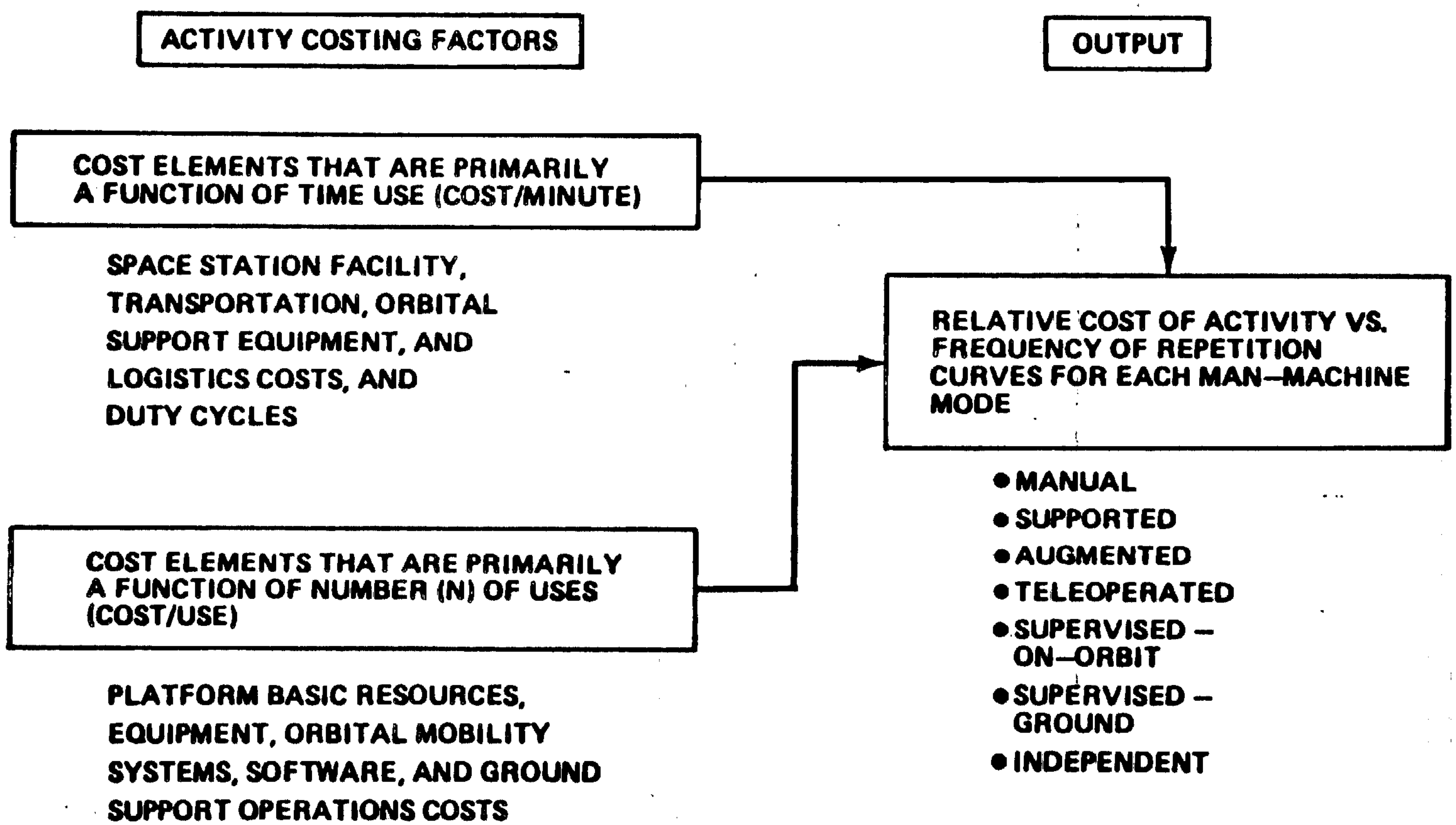
190-86

INSPECT/OBSERVE



181-86

THURIS COST METHOD



183-86

SUPPORT EQUIPMENT EVALUATION

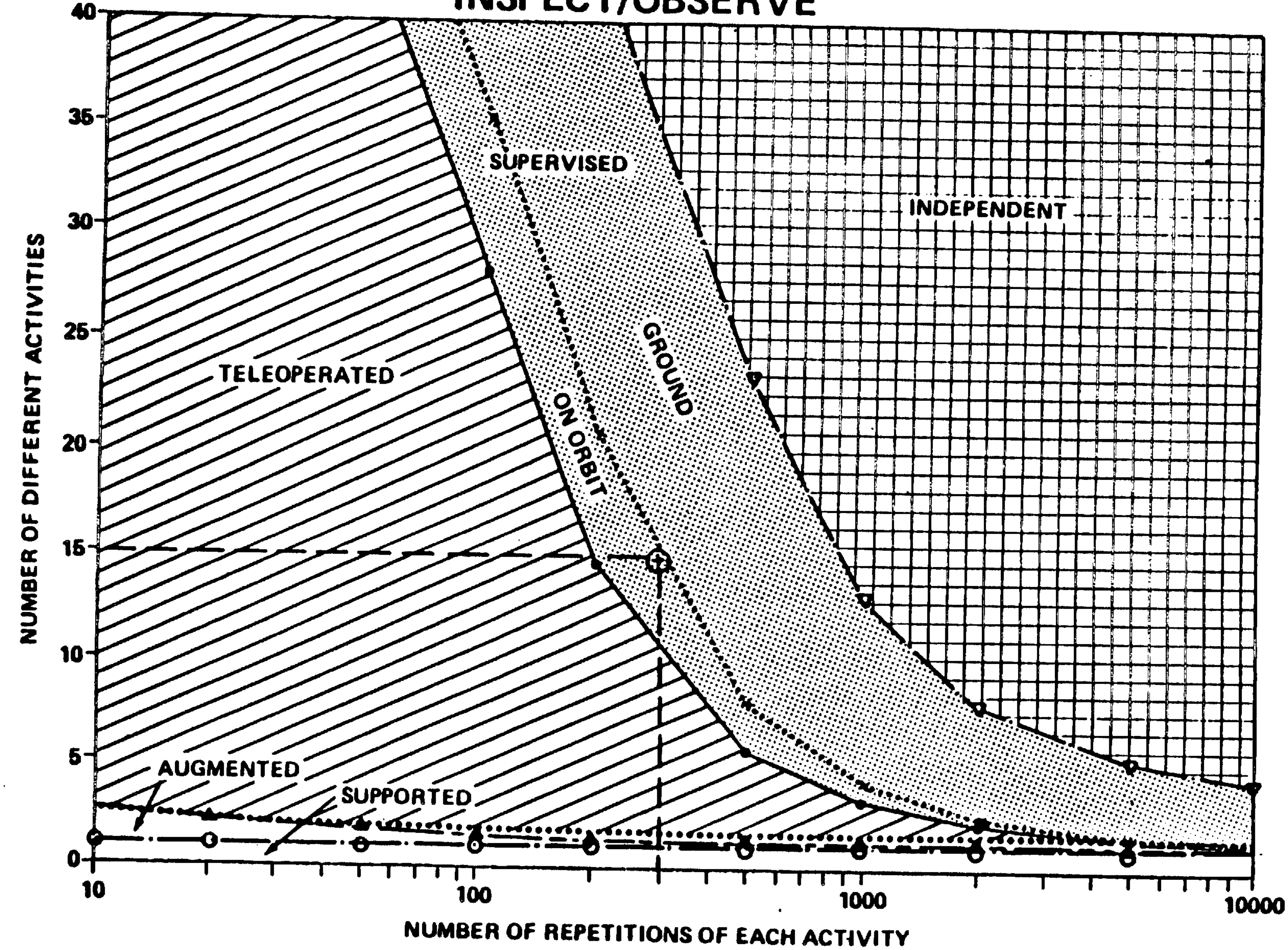
ACTIVITY NUMBER:	CATEGORIES OF MAN–MACHINE INTERACTIONS						
					SUPERVISED		
INSPECT/ OBSERVE	MANUAL	SUPPORTED	AUGMENTED	TELEOPERATED	GROUND	ON ORBIT	INDEPENDENT
	HUMAN SUPPORT EQUIP. REQUIRED						
IVA	A1	A1 C13	A1 C10 C13	A1	A2	A1	A2
EVA	A1 B1 C13	A1 B1 C13	A1 B1 C1[illegible] C1[illegible]				

EXAMPLE – INSPECT RETURNING SPACE BASED OTV

MANUAL	–	DIRECT VISUAL OBSERVATION
SUPPORTED	–	DIRECT OBSERVATION FROM FIXED WORK STATION
AUGMENTED	–	OBSERVATION THROUGH OPTICAL SYSTEM
TELEOPERATED	–	DIRECTLY CONTROLLED MANIPULATOR BASED TV
SUPERVISED GROUND	–	GROUND SUPERVISED TV
SUPERVISED ON–ORBIT	–	SPACE SUPERVISED TV
INDEPENDENT	–	PREPROGRAMMED TV SYSTEM WITH FEEDBACK

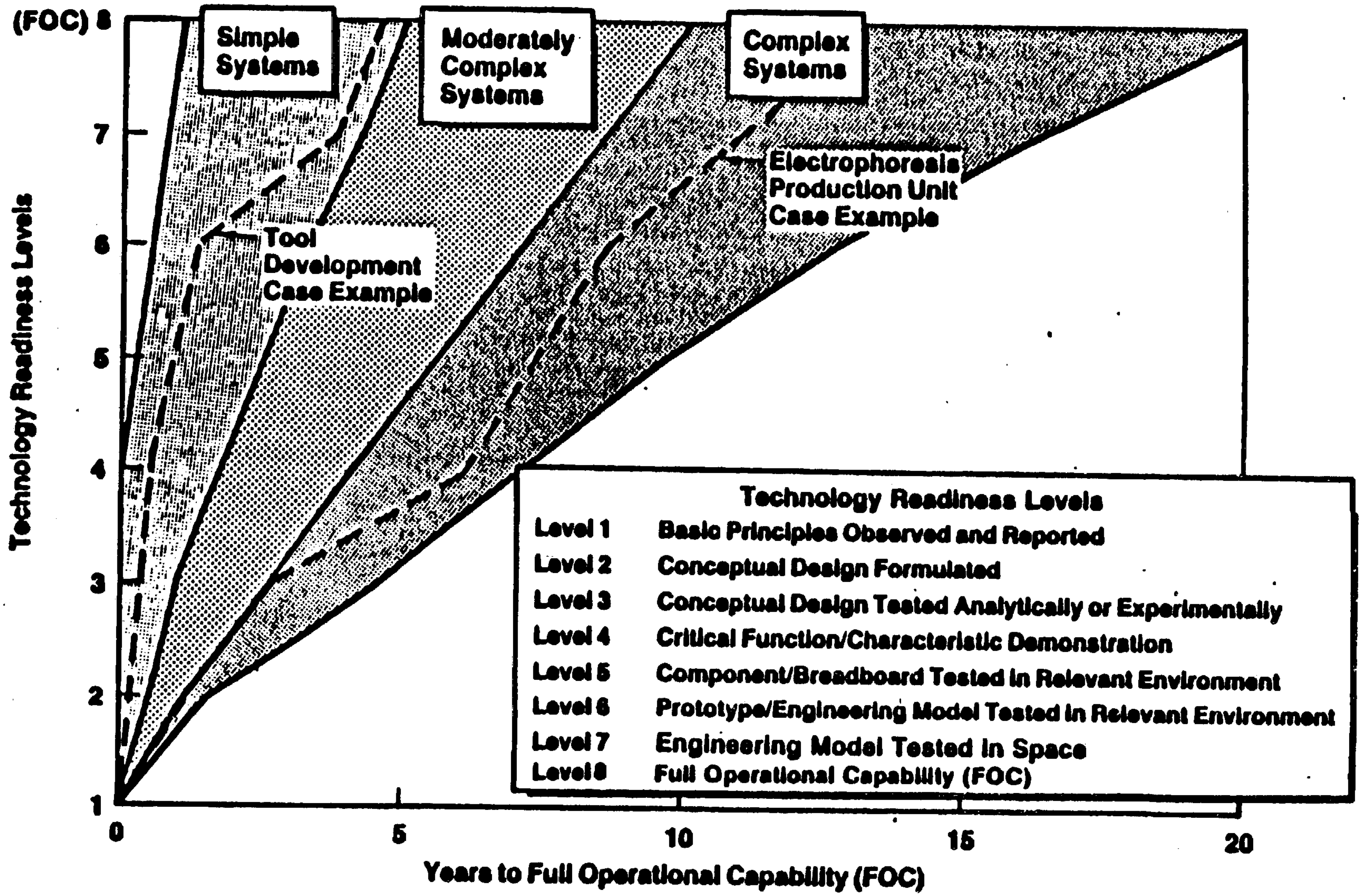
192-86

ACTIVITY NUMBER 22
INSPECT/OBSERVE



TECHNOLOGY ADVANCEMENT CLASSIFICATIONS

VGT266



573–86

THURIS – THE HUMAN ROLE IN SPACE

SUMMARY/CONCLUSIONS/RECOMMENDATIONS

- INTELLECT, SENSES, FINE MANIPULATIVE ABILITIES GOVERN HUMAN ROLES IN SPACE.
- DEVELOPED METHOD FOR SPACE ACTIVITY ALLOCATION BASED ON PERFORMANCE, COST, AND TECHNOLOGY READINESS CRITERIA.
- SMALL NUMBER OF GENERIC ACTIVITIES (37) DESCRIBE WIDE RANGE OF PROGRAMS.
- SEVERAL GENERIC ACTIVITIES (13) RELATED TO PROXIMITY OPERATIONS.
- TO ENHANCE HUMAN ROLES IN PROXIMITY OPERATIONS, TECHNOLOGY SHOULD EMPHASIZE:
  - PROJECTING PERFORMANCE POTENTIAL/SUPPORT REQUIREMENTS FOR ADVANCED MAN/MACHINE SYSTEMS.
  - FUNDAMENTAL UNDERSTANDING OF HUMAN INTELLIGENCE.
  - CONTINUING DEVELOPMENT OF ADVANCED VISUAL DISPLAYS.
  - VOICE INTERACTIVE SYSTEMS.

RENDEZVOUS AND PROXIMITY OPERATIONS WORKSHOP,
DISPLAYS AND HUMAN FACTORS SUBSESSION

Stereo Video and Display Systems

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Considerable interest has been shown in stereoscopic television systems which permit human operators to take advantage of their native ability to judge depth as a function of retinal disparity. The costs of such systems in terms of bandwidth, power, calibration and economics must be weighed against any increases in system performance when compared to a single camera monoscopic television system.

This presentation will discuss the simulation results comparing generic types of stereoscopic systems and will also address the human/machine issues of stereoscopic systems as they might be employed in proximity operations, especially those involved in docking or dexterious manipulation.

Engineering Approaches To Stereoscopic Television

The physiological fact that humans are capable of integrating two disparate visual inputs gives rise to our ability to sense depth over specific ranges. This capability can be reproduced through television systems by any of the following methods:

- Full Field Sequential Presentation - The display of a full field of video data to first one eye and then the other at rates exceeding Critical Fusion Frequency.
- Fuel Field Simultaneous Presentation - The display of a full field of video data to each eye. Each display is independent and presented to the two eyes through separate lenses.
- Split Field Presentations - The display of half a field of information to each eye through lenses either simultaneously or sequentially.

Candidate Stereoscopic Television Systems

Two stereoscopic television systems which have been evaluated for remote systems applications are the Fresnel stereoscopic display and the Piezoelectric Lens stereoscopic display.

The Fresnel system provides for simultaneous transmission of disparate images of the same scene to each eye. The Piezoelectric glasses provide high speed sequential presentation of two slightly different views of the scene. Both techniques give rise to the perception of depth of field.

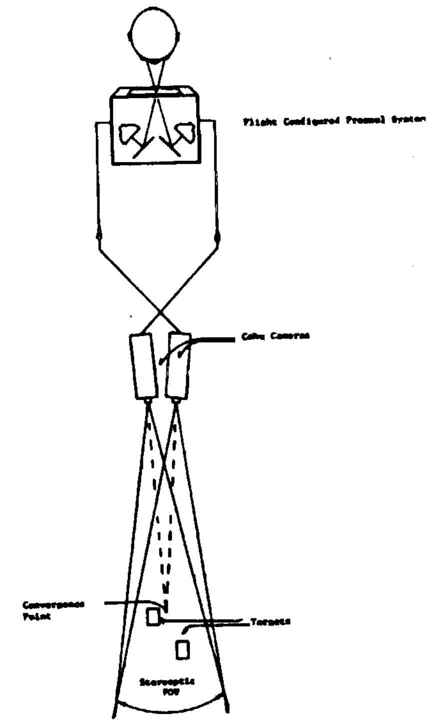


Figure 3-1: Flight Configured Fresnel Stereoscopic Display

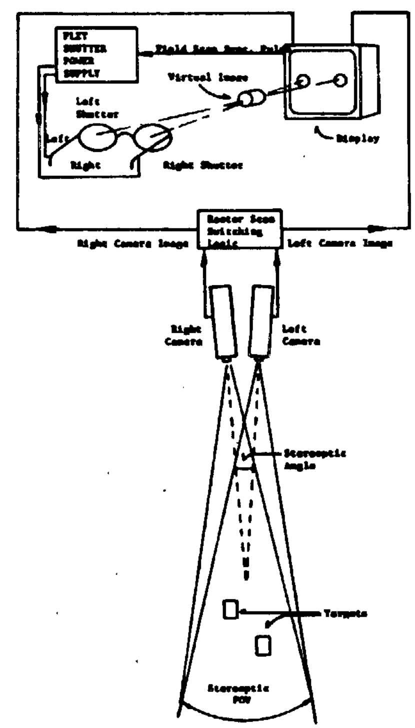


Figure 3-2: FLIT Stereoscopic Display

Areas of Applications

The candidate areas of operations for stereoscopic visual systems are:

1. Gaining ranging information from 50 meters to contact, with particular attention to ranging situations in complex backgrounds or docking situations which involve spacecraft appendages.

2. Close in docking maneuvers and fly-around inspection maneuvers.

3. Terminal grappling and docking execution.

4. Manipulation and manipulator tool use for remote servicing and maintenance.

Concepts of Applications for Stereoscopic Displays

While the retinal disparity provides steroscopic cues for humans, it is only one means by which we judge depth. In order to take full advantage by binocular stereopsis we can vary the two primary visual sensors in the following ways:

1. Manipulate the distance between the two sensors - This will have the effect of changing - usually increasing - the stereoscopic depth of field. Normal human stereo perception is lost beyond 20 feet, but by increasing the inter-camera distance, we can create stereo fields.

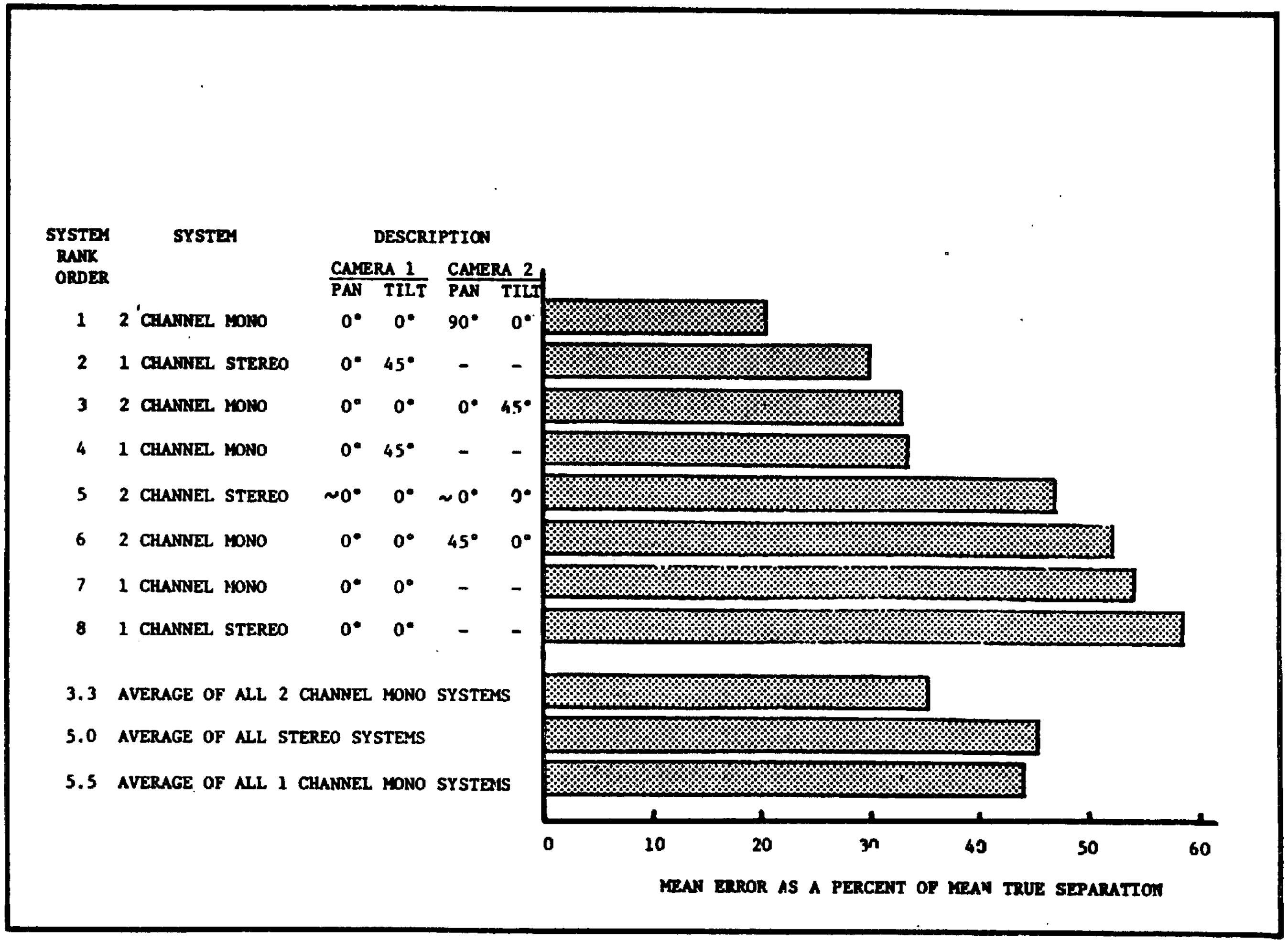
2. The cameras' convergence angle can be increased or decreased to enhance steroscopic conditions and provide the operator with exaggerated steroscopic cues, although there are some limits to this manipulation.

3. Additionally, monoscopic cues can be induced to enhance depth perception. These would include use of on-board lighting to proxide illumination differences and camera movements to provide interpositional cues and perspective.

Effects of Stereoscopic Visual Systems on System Performance

Logically, stereoscopic television systems should offer some advantages in displaying depth in a visual field over single camera monoscopic systems.

This supposes, however, that cameras must be arranged as our eyes are in our head - laterally offset and in plane. In engineering video sensors it might be possible to arrange them otherwise and further enhance depth cues.



Comparison Among Camera System Types and Geometric Arrangements used in a Two Target Fore-Aft Separation Judgement Task

Future Requirements

Newer stereoscopic television systems are currently being integrated into the MSFC Teleoperator and Robotics Evaluation Facility and will be subjected to system level evaluations. In conjunction with stereoscopic systems, other ranging and depth cues will be studied such as reticles, independent monoscopic cameras, radars, proximity sensors and specialized lighting.

Conclusions

- Stereoscopic vision systems offer general performance advantages over a monoscopic system.
- Bandwidth and power requirements are increased while luminance of the display decreases when using stero systems.
- There are some minor restrictions on head movement with state-of-the art stereoscopic displays either due to the display (Fresnel) or head mounted equipment (Piezoelectric).
- The option to employ steroscopic vision should be considered in view of task tolerances, task criticality system simulation results which explore a full range of visual system alternatives.

27 MARCH 1985

SESSION 8 - SYSTEMS INTEGRATION AND DEVELOPMENT SUPPORT

8-1. "OMV UTILIZATION FOR LARGE OBSERVATORY MISSION SUPPORT" - FRANK SWALLEY/NASA MSFC

8-2. "INTERACTION BETWEEN THE SPACE TRANSPORTATION SYSTEMS AND THE SPACE STATION ARCHITECTURE AND OPERATIONS AND TECHNOLOGY REQUIREMENTS" - DONALD EIDE/NASA LARC

8-3. "SYSTEMS INTEGRATION METHODOLOGIES" - FRANK VINZ/NASA MSFC

8-4. "CREW SYSTEMS, SPACE TELESCOPE MAINTENANCE AND REFURBISHMENT" - JOHN REAVES/NASA MSFC

8-5. "SYSTEM INTEGRATION/SIMULATION TESTS" - A. NATHAN/GRUMMAN AEROSPACE CORPORATION

8-6. "RENDEZVOUS AND DOCKING TECHNOLOGY DEVELOPMENT FOR FUTURE EUROPEAN MISSIONS" - WIGBERT FEHSE/EUROPEAN SPACE AGENCY, ESTEC

8-7. "DEMONSTRATION MISSION FOR AUTONOMOUS RENDEZVOUS WITH THE EURECA PLATFORM" - B. CLAUDINON/MATRA ESPACE, P. NATENBRUK/MBB-ERNO, AND H. P. NGUYEN/AEROSPATIALE

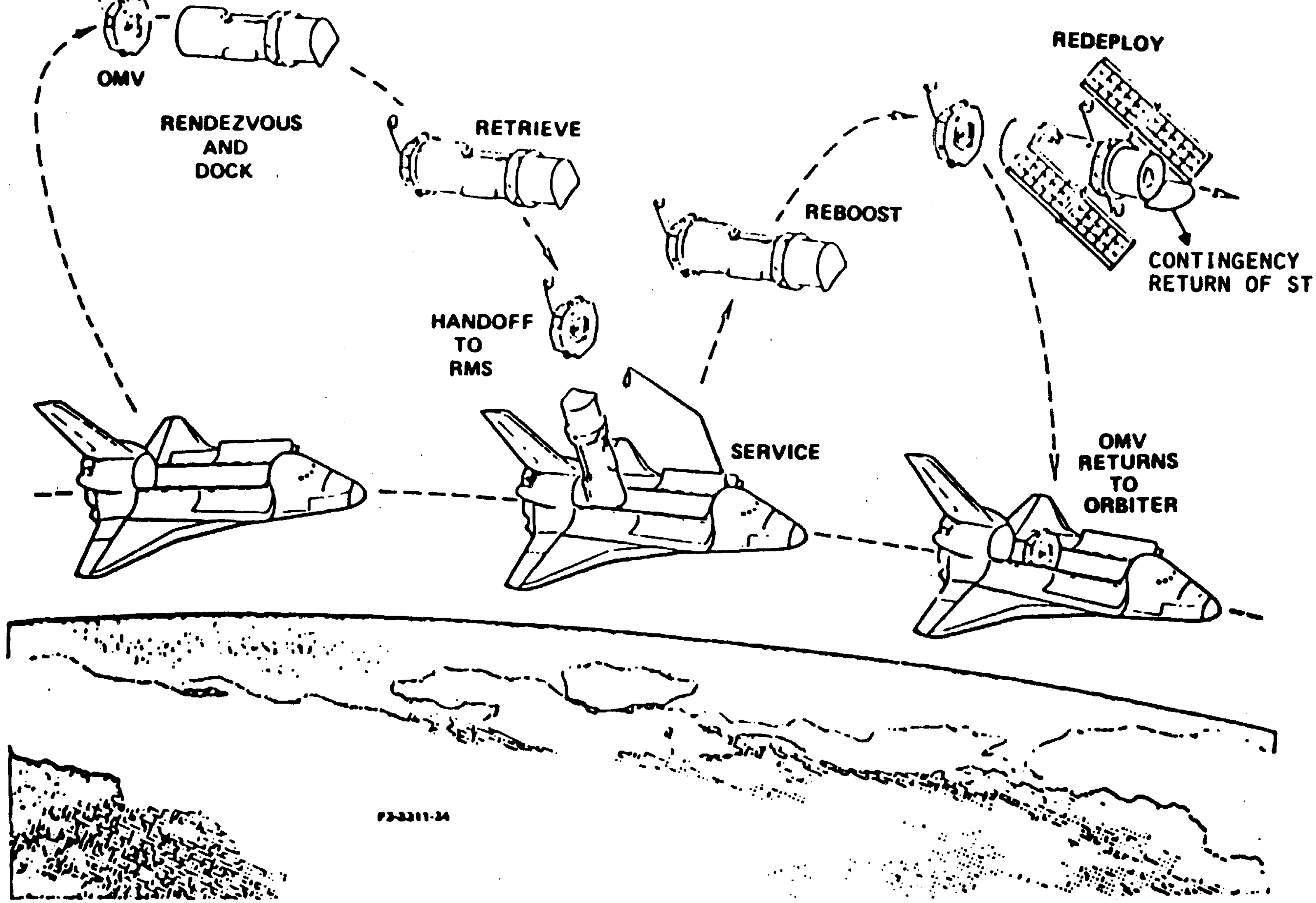
574–85

OMV UTILIZATION FOR LARGE OBSERVATORY MISSION SUPPORT

- OVERVIEW OF MISSIONS
- MISSIONS REQUIREMENTS
  - OMV/ST DOCKING MECHANISMS
- SUMMARY OF REQUIREMENTS ON THE OMV
  - MISSION PROFILES
- SUMMARY OF DESIGN CONSIDERATIONS
  - OMV
  - OBSERVATORY

3747-83

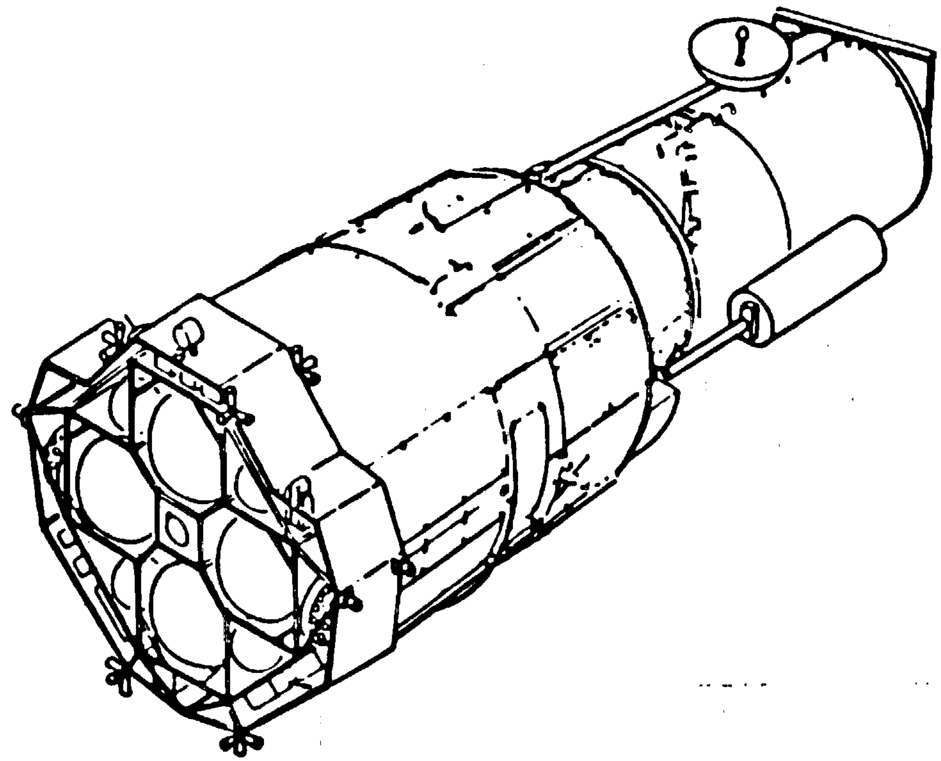
RETRIEVAL, SERVICING AND REDEPLOYMENT WITH OMV



- DETAIL TIMELINES, ETC. WILL BE SHOWN ONLY FOR THE "DRIVER MISSIONS" E.G. INITIAL PLACEMENT NOT SHOWN

563–85

OMV/ST DOCKING MECHANISMS



DOCK TO AFT BULKHEAD OF THE SPACE TELESCOPE

TO ALLOW DOCKING TO ST AFT BULKHEAD

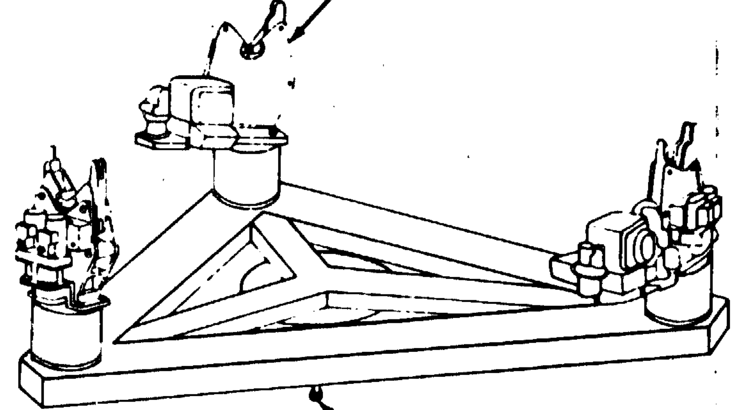
- REPLACE RMS/EE ON THE OMV WITH 3 FSS LATCH ASSEMBLIES MOUNTED TO A SUPPORT RING

SUPPORT RING

- CONCERN ABOUT THE ABILITY OF OMV AND ST TO ACCOMPLISH SUCH A "PLANAR" DOCKING
  - S&E SIMULATIONS ARE PLANNED TO VERIFY

TO ALLOW DOCKING OF OMV TO ST AFT BULKHEAD

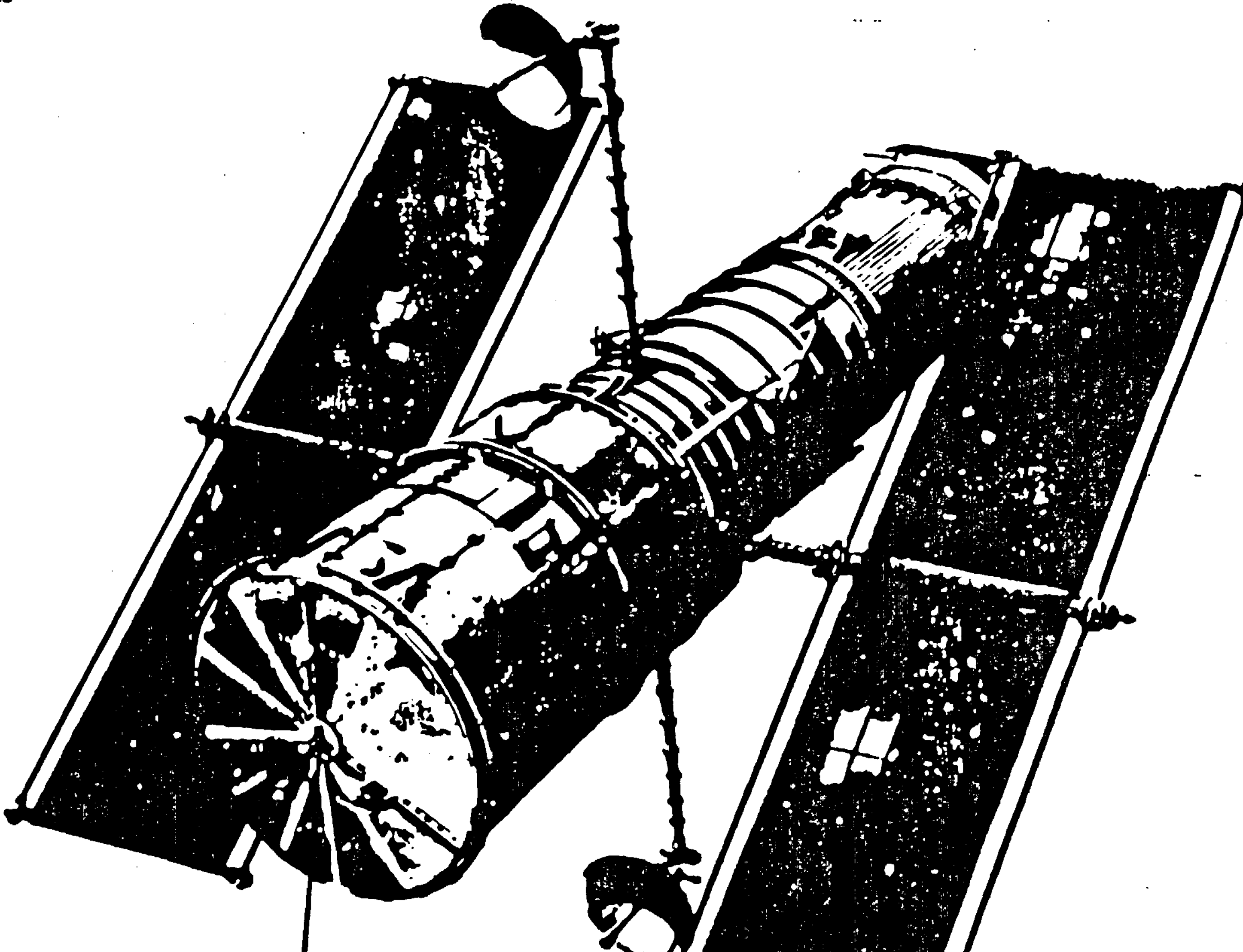
- MOUNT AN ADAPTER ASSEMBLY TO THE ST AFT BULKHEAD



- ADAPTER ASSEMBLY WOULD HAVE TO BE ATTACHED AND REMOVED FROM THE ST BY EVA

566–85

TO ALLOW DOCKING TO ST AFT BULKHEAD



567–85

ST/OMV SERVICING MISSION
SUMMARY OF REQUIREMENTS ON ST

- NO IMPACTS, IF OMV CAN DOCK DIRECTLY TO ST USING FSS LATCH ASSEMBLIES

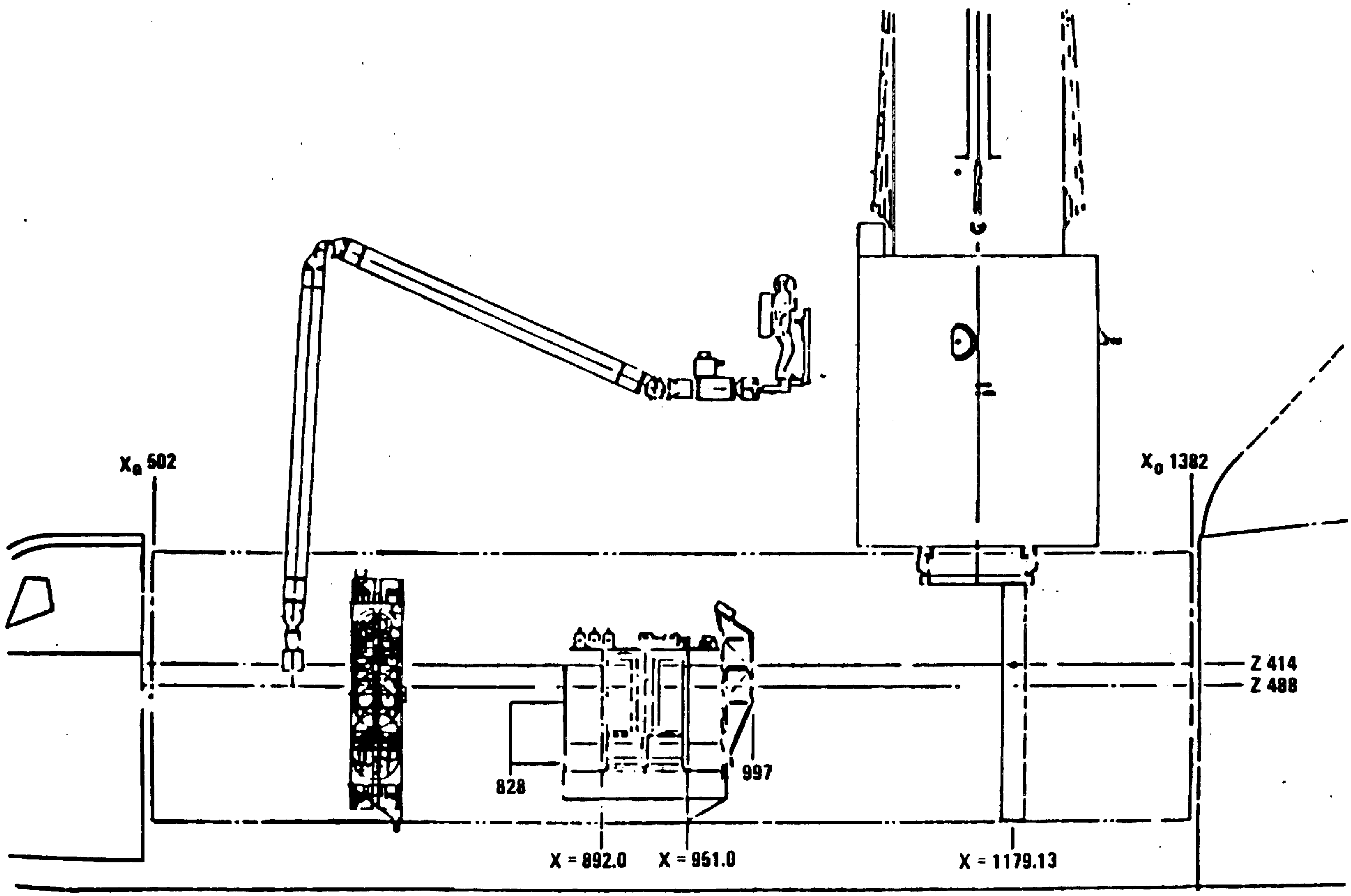
- MAY HAVE TO ADD AN RMS GRAPPLE FIXTURE TO THE CENTER OF THE ST AFT BULKHEAD

OR

- MAY HAVE TO HAVE AN ADAPTER ASSEMBLY ATTACHED AND DETACHED TO THE FLIGHT SUPPORT SYSTEM (FSS) PINS ON THE ST AFT BULKHEAD BY EVA

568–85

MAINTENANCE CONFIGURATION



505-34

ST/OMV SERVICING MISSION
SUMMARY OF REQUIREMENTS ON THE OMV

AREA	ST SOLAR ARRAYS IN	ST SOLAR ARRAYS OUT (NO POWER DURING THRUSTING)	ST ARRAYS OUT (CONTINUOUS POWER)
MISSION	RETRIEVE REDEPLOY NO ELEC POWER TO ST	REDEPLOY PLUS CONTINGENCY RETURN	REDEPLOY PLUS CONTINGENCY RETURN
MISSION DURATION (HRS)	RETRIEVE –27 REDEPLOY – 12	55.8 HRS	55.8 HRS
BATTERIES	5	23 (2)	DURING OMV THRUSTING 17 (HOLD OMV/ST ROLL ANGLE) 11 (FULL ARTICULATION OF ARRAY)
OMV PROPELLANT	LESS THAN 6700 LBS	4000 (1)	
ENGINES	USE MAIN THRUSTERS	3 BURNS AT 3 HRS (4) EACH (40 LB TOTAL THRUST)	
CONTROL SYSTEM	MAIN THRUSTERS – BANG OFF RCS – ROLL CONTROL	MINIMIZE LOADS TO (3) EXTENDED ARRAYS	

OMV IMPACTS:

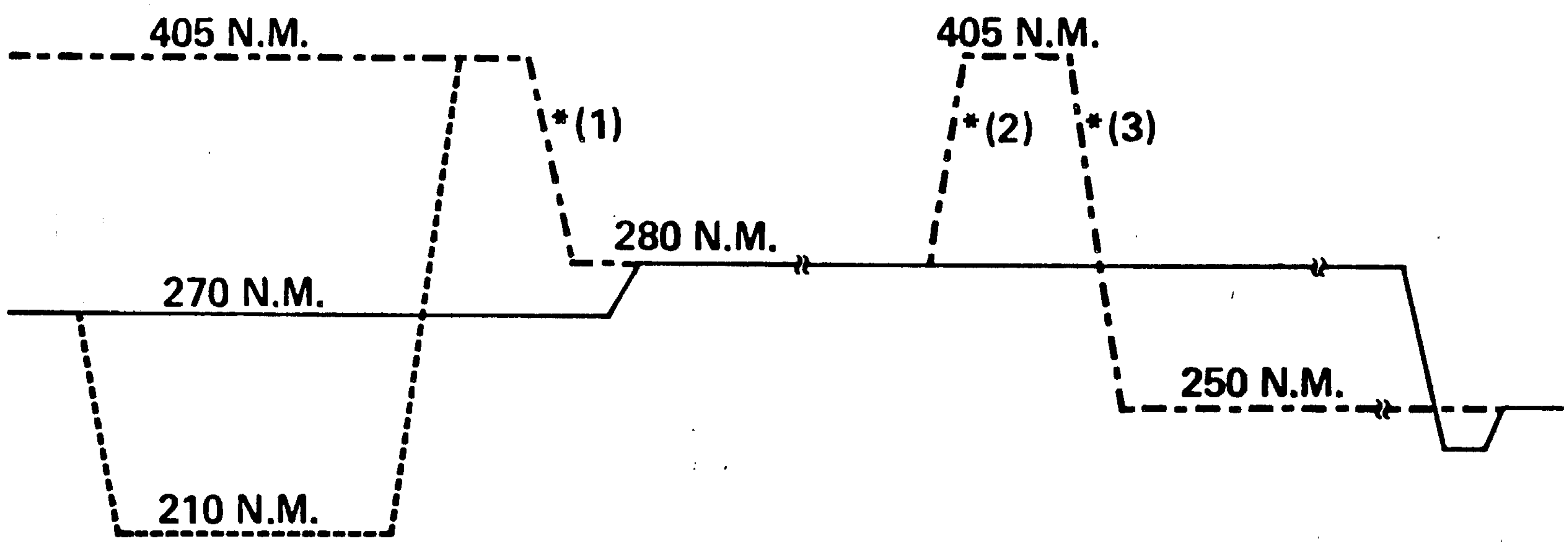
(1) INCREASE OMV TANKAGE SIZE OR ADD A PROPELLANT KIT OR REFUEL AT THE SHUTTLE
(2) ADD BATTERIES AND ACTIVE THERMAL CONTROL SYSTEM FOR THEM
(3) MODULATE CONTROL ENGINE THRUST, OR GIMBAL CONTROL ENGINES, ETC.
(4) QUALIFY ENGINES FOR LONG BURNS

570–85

- CONTINGENCY RETURN
- 1700 WATTS ELECTRICAL POWER TO ST
- SOLAR ARRAYS DEPLOYED

RETRIEVAL

DEPLOYMENT/CONTINGENCY RETURN



——— ORBITER
- - - - ST (+ OMV)
-------- OMV

OMV IMPACTS

- BURN TIME *(1) 2.46 HR.
  *(2) 2.81 HR.
  *(3) 3.20 HR.
- BATTERIES – 23–11
- OMV PROPELLANT – 7500 LB. ADEQUATE

571–85

DESIGN CONSIDERATIONS FOR SUPPORTING LARGE OBSERVATORY MISSIONS

OMV

- LONG BURN TIMES (3 HRS) AT LOW THRUST LEVELS (40 LB TOTAL THRUST)
- LONGER MISSION DURATIONS RE–QUIRING
  - POWER
  - THERMAL CONTROL
- CONTROL SYSTEM TO MINIMIZE THE DYNAMICS ON THE OBSERVATORY'S EXTENDED SOLAR ARRAYS

OBSERVATORY

- STRENGTHENED SOLAR ARRAY ATTACHMENT OR HIGH RELIABILITY EXTENSION AND RETRACTION MECHANISMS FOR SOLAR ARRAYS
- PROVISIONS FOR GRAPPLING FIX–TURES FOR OMV

INTERACTION BETWEEN THE SPACE TRANSPORTATION SYSTEMS AND THE SPACE STATION ARCHITECTURE AND OPERATIONS AND TECHNOLOGY REQUIREMENTS

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Rendezvous and Proximity Operations Workshop
Johnson Space Center
Houston, Texas
February 19-22, 1985

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ABSTRACT

This study, limited to the interaction between the Space Station and an orbital transfer vehicle, illustrates the need to include the space transportation system and the station in a synergistic analysis to determine an overall operationally efficient system that takes full advantage of the potential for the exploration and exploitation of space. Included are examples of the effects of the mission model on the selection of a possible orbit transfer vehicle design and processing procedures at the station; safe, timely, and flexible transfer vehicle trajectories from GEO to the station; mechanical assists on the station for vehicle arrivals and departures that reduce the risk of contamination and collision; and a discussion of the need for efficient procedures for vehicle maintenance and processing at the station.

INTRODUCTION

The advent of the Space Shuttle has opened the doors to many, heretofore, unavailable opportunities for the exploration and exploitation of space. These opportunities will be greatly expanded by the addition of a Space Station. Greater and broader capabilities will be achieved with the addition of a space-based transportation system that is both an extension of Space Station capabilities and a user of time and resources from the station. All these transportation elements, when combined, will form a complete space system that will require operational efficiency for success.

Many studies of the technology requirements and operations of the Space Station and of the transportation system have been conducted in the past, but few included both of these key elements. There is a synergism between these elements that extends from the ground to geosychronous orbit (GEO) and beyond that requires efficient procedures and technologies for maximum utilization of the transportation system and the Space Station. Analysis without consideration of the total system can result in a lack of identification of many important issues and problems.

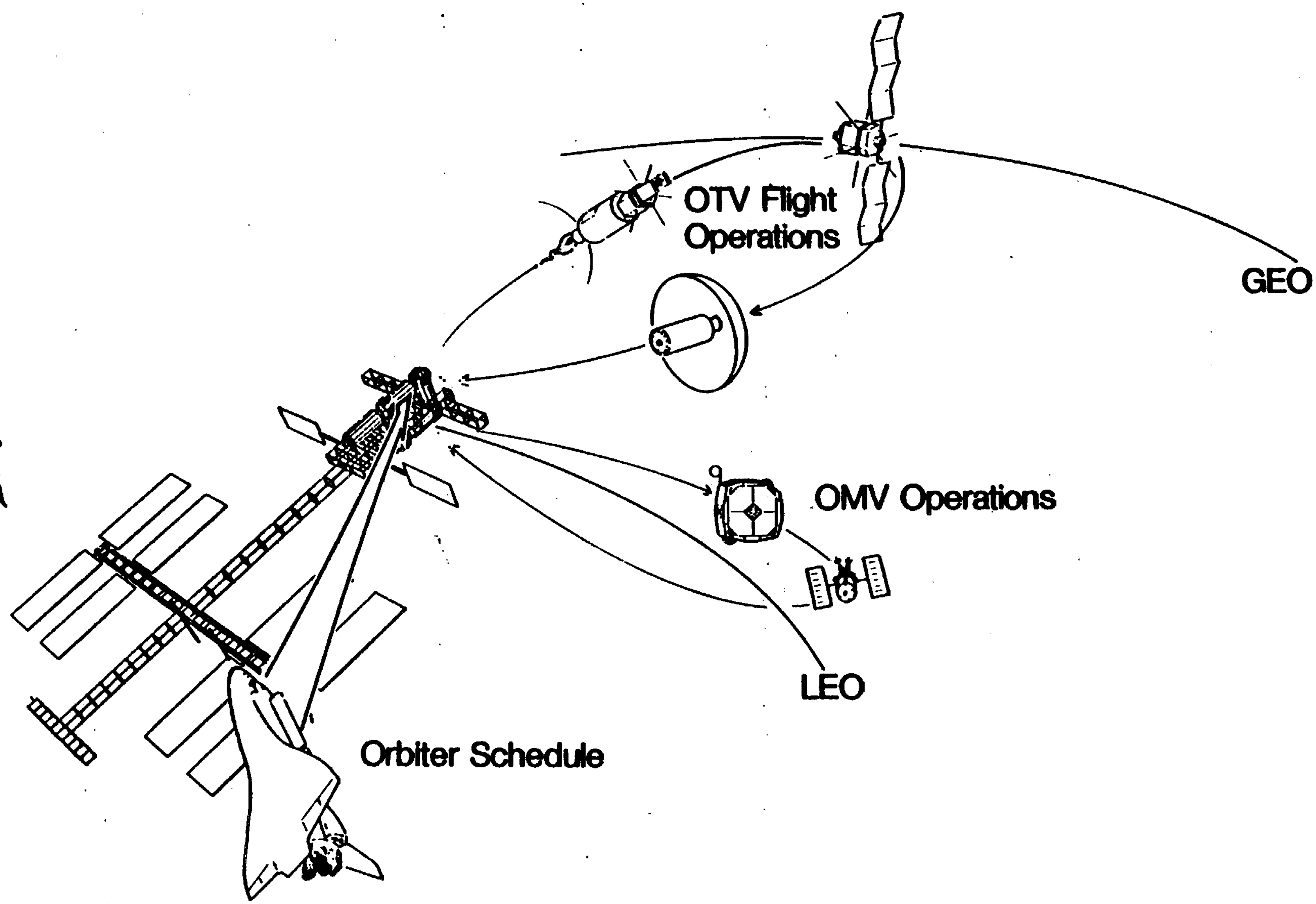


Figure 1. A Complete Space System

SPACE TRANSPORTATION SYSTEM - SPACE STATION SYNERGISM

Analyses to determine designs and operations of the transportation system and Space Station that perform efficiently as a complete space system require consideration of the synergism between the Space Station and the space transportation system. The demands placed on the Space Station by the space transportation system are illustrated in Figure 2. These demands include facilities such as propellant storage tanks, hangers for maintenance and processing of the orbital transport vehicles and orbit maneuvering vehicles (OMV), critical-spares storage. tracking. and monitoring of the transportation system. Another demand is for Space Station crew time to perform the maintenance, processing, and monitoring of the transportation system.

As the number of arrivals and departures at the Space Station increases, the total number of facilities and the total amount of Space Station crew time required to support the space transportation system increase. These increases will produce pressures for procedures, safe trajectories and maneuvers, and technologies that minimize the demands for additional resources. Minimization of these demands requires an analysis that includes the space transportation system concepts and designs and the Space Station architecture and constraints.

This preliminary analysis was limited to the interaction between the Space Station and an orbital transfer vehicle to illustrate the need for a synergistic analysis and identify some issues and problems. The analysis included the effects of the mission model on selection of an orbital transfer vehicle design and processing procedures at the Space Station; trajectories that provide safe, timely, and flexible approaches to the station from GEO; a determination of the need for mechanical assists on the station that reduce the risk of contamination and collision with the station during vehicle arrival; and a discussion of the need for analyses of vehicle processing procedures at the station to identify technology improvements and procedures for efficient oprations.

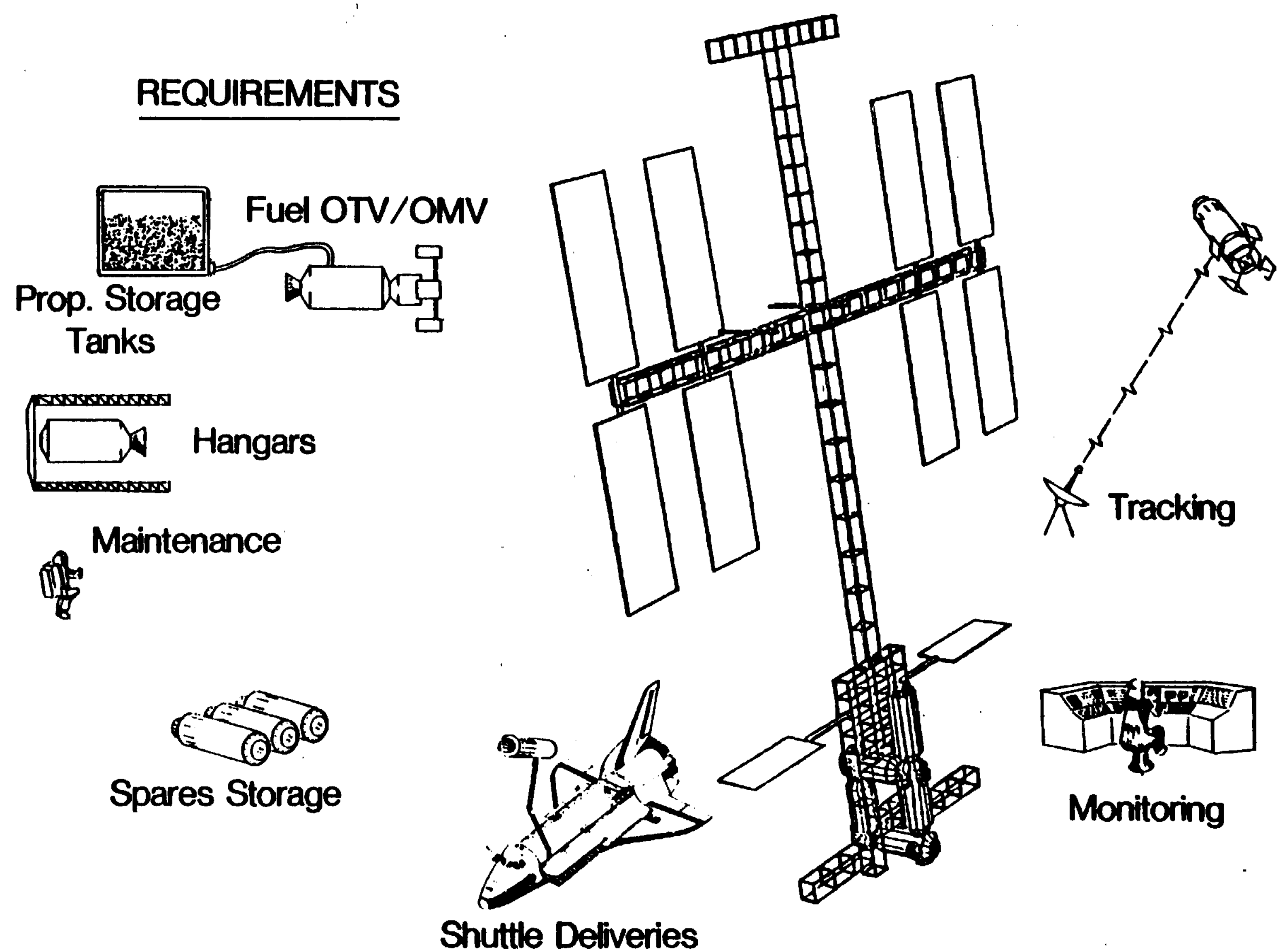


Figure 2. STS - Space Station Synergism

MISSION MODEL

A major driver in the determination of time and resource support required by the transportation system at the Space Station is the mission model. A 10-year mission model assumed for this study, starting approximately in 1994, was a modified MSFC model (Revision 7) consisting of one-way deliveries of 236 payloads to GEO by a reusable space-based orbit-to-orbit transportation system. This mission model serves to demonstrate the need for this type of analysis. Although not a large portion of this mission model, significant numbers of missions from the station to refurbish satellites in GEO can increase the need for either life support modules, teleoperators, or robotics on these transport vehicles for in-situ satellite repair and/or refurbishment.

The cumulative distribution of payload weight to be delivered to GEO each year for the total 10-year mission model assumed in this study is shown on Figure 3. The annual distributions are dominated by payloads that are 3000 pounds and less and indicate a desire for an orbital transport vehicle designed for small payload deliveries. However, examination of the cumulative distribution of payload sizes for the entire 10-year program shows that a transport vehicle designed to deliver a 10,000-lb payload to GEO can capture 93 percent of all these GEO missions. In this study, it was assumed that, for efficiency, up to four payloads could be prepackaged into approximately 5000-lb cargos for delivery to GEO in a single launch. This multiple manifesting reduced the 236 payloads to 91 missions that included 14 missions that weighed between 10,000 and 20,000 lbs.

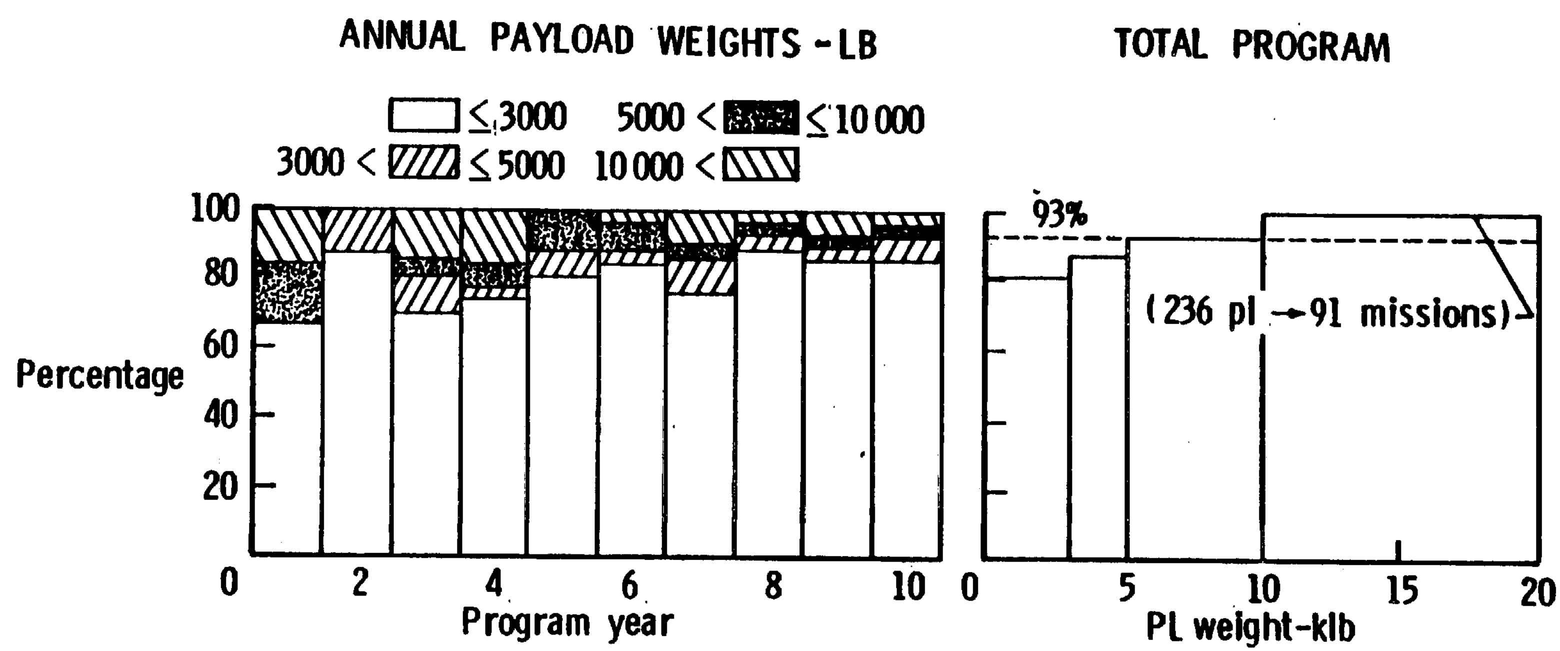


Figure 3. Payload Weight Distributions
(Orbital Transfer Vehicle Delivery to GEO)

TRANSFER VEHICLE DESIGNS

The preliminary designs obtained with AVID for two different space-based concepts configured to deliver 10,000 lbs of cargo one way to GEO are shown on Figure 4. These concepts included an all propulsive orbital transfer vehicle (OTV) and an aeroassisted orbital transfer vehicle (AOTV). These concepts were equipped with "intelligent end effectors" for automated cargo mating and alignment with the transport vehicle c.g. prior to launch. These end effectors were also assumed to work in conjunction with an adaptive guidance and control system onboard the vehicle to keep the cargo plus vehicle c.g. within a prescribed envelope. This adaptive system would be required because of the expected wide range of cargo sizes and shapes. Both concepts have composite structures that in previous studies (NASA CR 3766) have been shown to provide reasonable efficient delivery when propellant is off loaded for lighter-weight cargo. They also use aluminum tanks and a LOX/LH_2 propulsion system with a baseline Isp = 459 sec that is representative of current technology (RL 10). There are other concepts for an AOTV that could be used, but this one was selected for illustrative purposes and does not imply that this concept is perferred. It was assumed that payloads heavier than 10,000 pounds would use additional stages (expendable or reusable) for the needed performance.

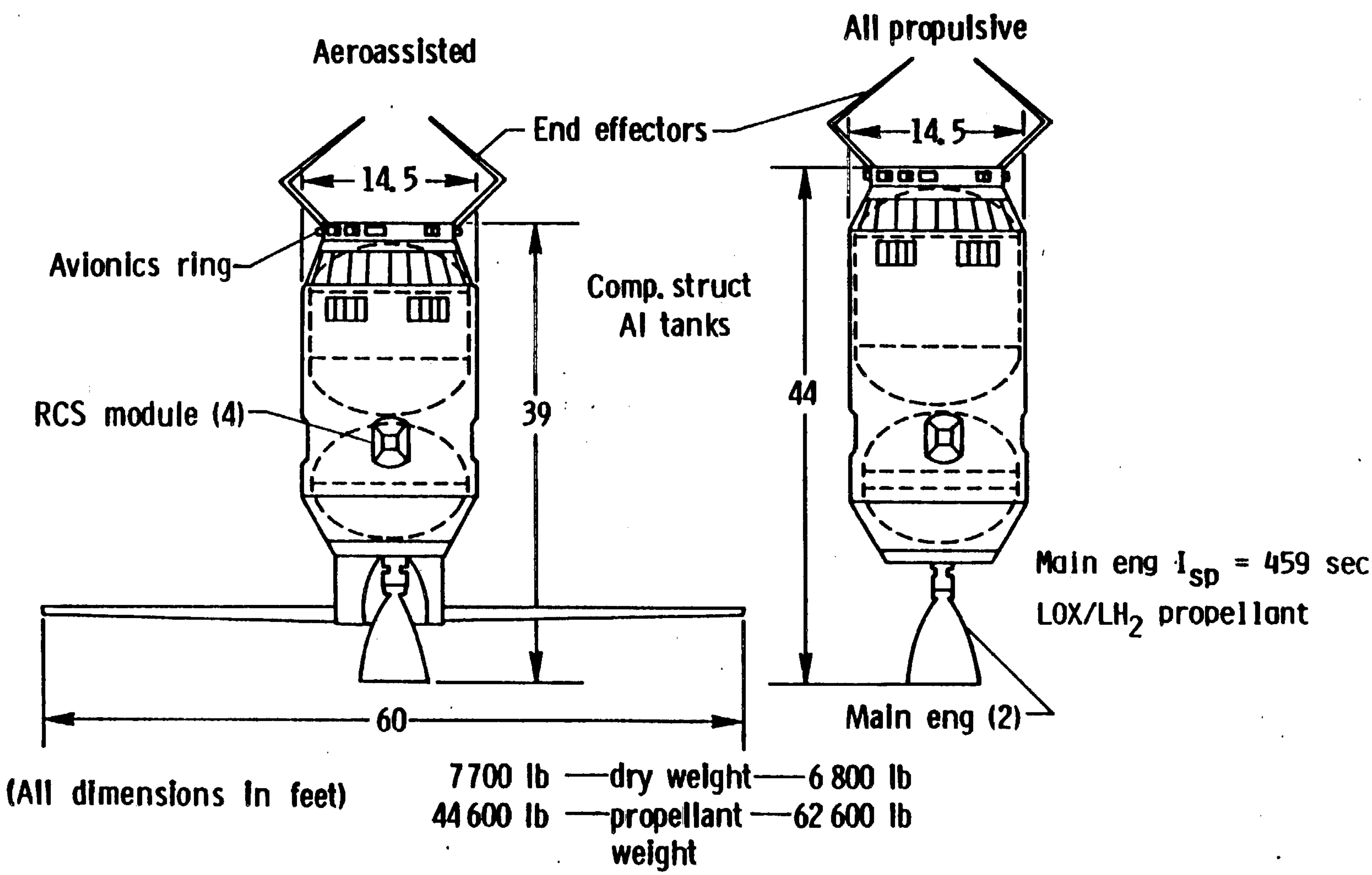


Figure 4. Space Based Orbital Transfer Vehicle Designs

ORBIT TRANSFER VEHICLE
PROPELLANT CONSUMPTION PER FLIGHT COMPARISONS

The maximum propellant consumption per flight required by the two OTV concepts with the baseline Isp of 459 and an improved Isp of 482 are compared on Table I for 20,000 and 10,000 lbs of payload. The differences in consumption have implications beyond delivery cost. Reduced propellant consumption reduces on-orbit storage requirements on or near the station which can impact station design and operation. However, there is an option that may reduce some of the impact. On Table I note that the maximum propellant consumption per flight required by the AOTV, with or without an improved Isp, is approximately equal to the projected direct launch capability to the station by future orbiters. If the maximum AOTV (or OTV) delivery capability to GEO were to be determined by the maximum propellant delivery capability to the station by a future or current orbiter, the on-orbit propellant storage problem could be significantly reduced by utilizing direct on-orbit fueling from the orbiter to the transfer vehicle. This option may best be exercised early in the AOTV involvement in the program until the mission requirements are more clearly established. Some on-orbit propellant storage would be required for flexibility and quick response missions.

The technological uncertainties that affect the viability of this scheme include propellant boiloff in zero g, propellant storage location, and delivery options.

Orbit transfer vehicle concept	Max propellant per flight-lb	
	Max vehicle pl capability	
	20 000 lb	10 000 lb
All propulsive		
$I_{sp} = 459$ sec	85 800	62 500
$I_{sp} = 482$ sec	74 500	54 000
Aeroassisted		
$I_{sp} = 459$ sec	64 200	44 500
$I_{sp} = 482$ sec	57 900	39 500

Table I. Orbital Transfer Vehicle Propellant Per-Flight Consumption Comparisons (LOX/LH2)

COMPARISONS OF ANNUAL PROPELLANT CONSUMPTION

The selection of the payload weight delivery capability for a transport vehicle concept, multiple manifesting, and the technology improvements to be incorporated in the design can significantly impact propellant consumption and the overall operation of the vehicle and station. This impact is illustrated by the comparison on Figure 5 of annual propellant consumptions for the AOTV and OTV concepts designed to deliver a 10,000-lb payload to geosynchronous orbit with additional kick stages (expendable or reusable) used for heavier payloads. These concepts utilize an engine with an Isp equal to either a baseline 459 sec or an improved 482 sec. A larger reduction in propellant consumption can be achieved with an AOTV than for an OTV concept that uses an improved Isp. Incorporating an improved Isp into an AOTV design provides an additional, but smaller, percent reduction in propellant consumption.

Depending on the number of missions, the differences in propellant consumption result in reductions of annual propellant usage requirements of almost 200,000 lbs for the different concepts. Propellant losses from on-orbit storage would increase this number. Without definitive knowledge of future missions, it would be difficult to determine the size, number, or type of storage tank to design for the future. Although there are different options for procedures to deliver propellant (e.g. tankers, aft cargo carriers, scavenging . . .), it remains uncertain if the storage tanks should be placed on the station or kept at a separate depot. Propellant storage on the station would have problems associated with large variations in mass and sloshing. A separate depot requires tending, station keeping, and transits to and from the station during vehicle processing.

Overall scheduling of orbiter deliveries to the station with and without propellant can impact the number of vehicles in the fleet and may require more flexibility in scheduling. The flexibility can be improved by locating the station in a rendezvous compatible orbit that provides two opportunities a day to launch to the station. These scheduling scenarios require analysis of the complete process including the potential use of an OMV as an intermediate stage.

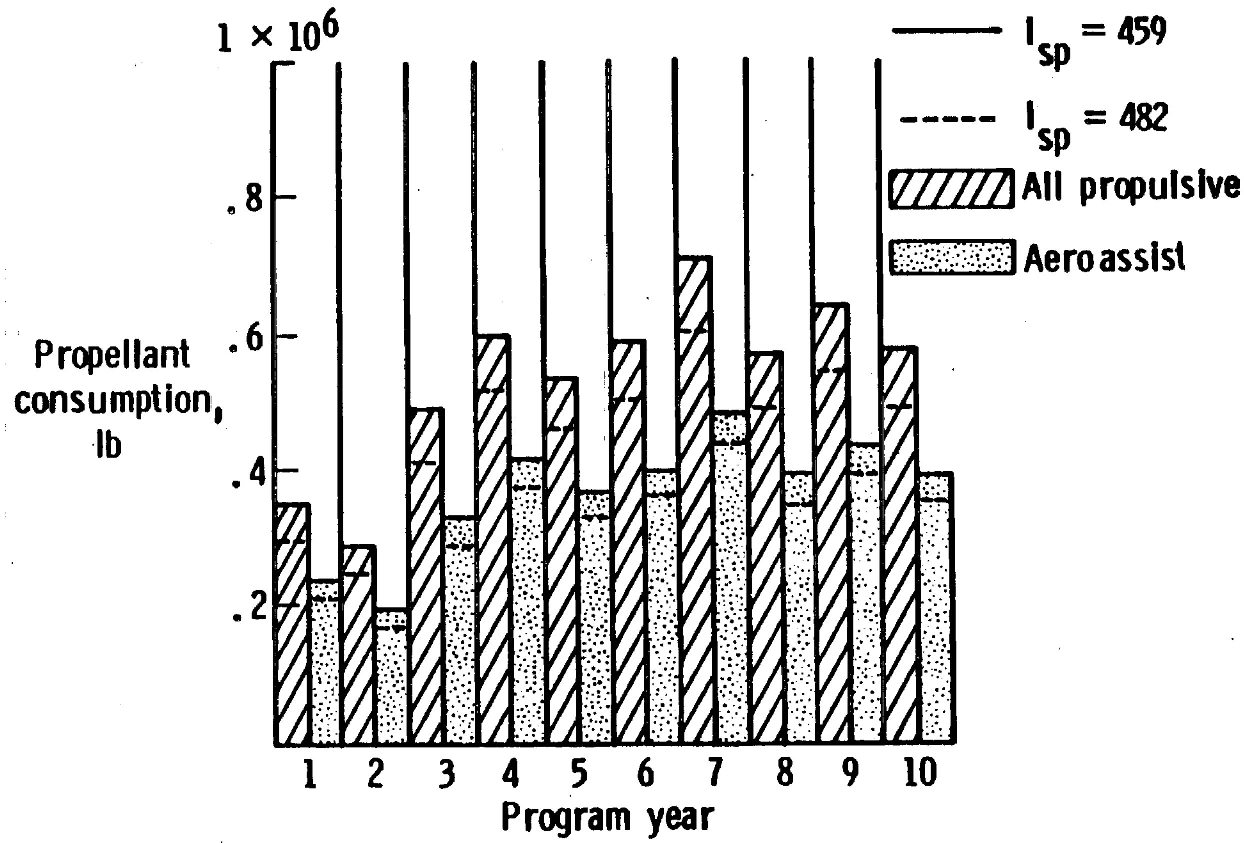


Figure 5. Comparisons of Annual Propellant Consumption (10K Pl. Delivery + Kick Stage)

GEO TO LEO TRANSFER
(Phasing Times to Reach Hohmann Tranfer Points)

Another important part of flight operations is the scheduling of missions from the station to GEO and return. These missions require careful scheduling and planning to reach the desired final location with a minimum expenditure of fuel and trip time. The return from GEO requires coordination between ground and station control to also minimize fuel expenditure and trip time. If there is a lack of coordination and ground control commands the transport vehicle to an unsuitable position on a parking orbit within the station control zone, excessive amounts of time can be required waiting for the transport vehicle to reach optimum Hohmann transfer points on the orbit.

The waiting (phasing) times required for different parking orbits above the Space Station are illustrated on Figure 6 for range angles measured clockwise from the orbit transfer vehicle to the station. For example, if the transfer vehicle is placed on an intermediate circular parking orbit 10 n.mi. above the Space Station with a relative angle 90° ahead of the Space Station, a phasing time of about 4 days is required to reach the minimum transfer velocity points. Initial positions below and behind the station produce a mirror image of this figure. The amount of time spent tracking and monitoring the transport vehicle can prove burdensome to the station, result in an inefficient use of resources, and produce a backlog of missions if the total number of missions increase beyond prescribed levels.

Initiating the rendezvous maneuver from non-optimum points on a parking orbit can probably be expected because of potential conflicts between the arriving vehicle and unscheduled Space Station activities and a desire for flexibility in planning. The penalities for non-optimum rendezvous maneuvers need to be examined to establish reasonable zones for safe, timely, and reasonably efficient rendezvous maneuvers.

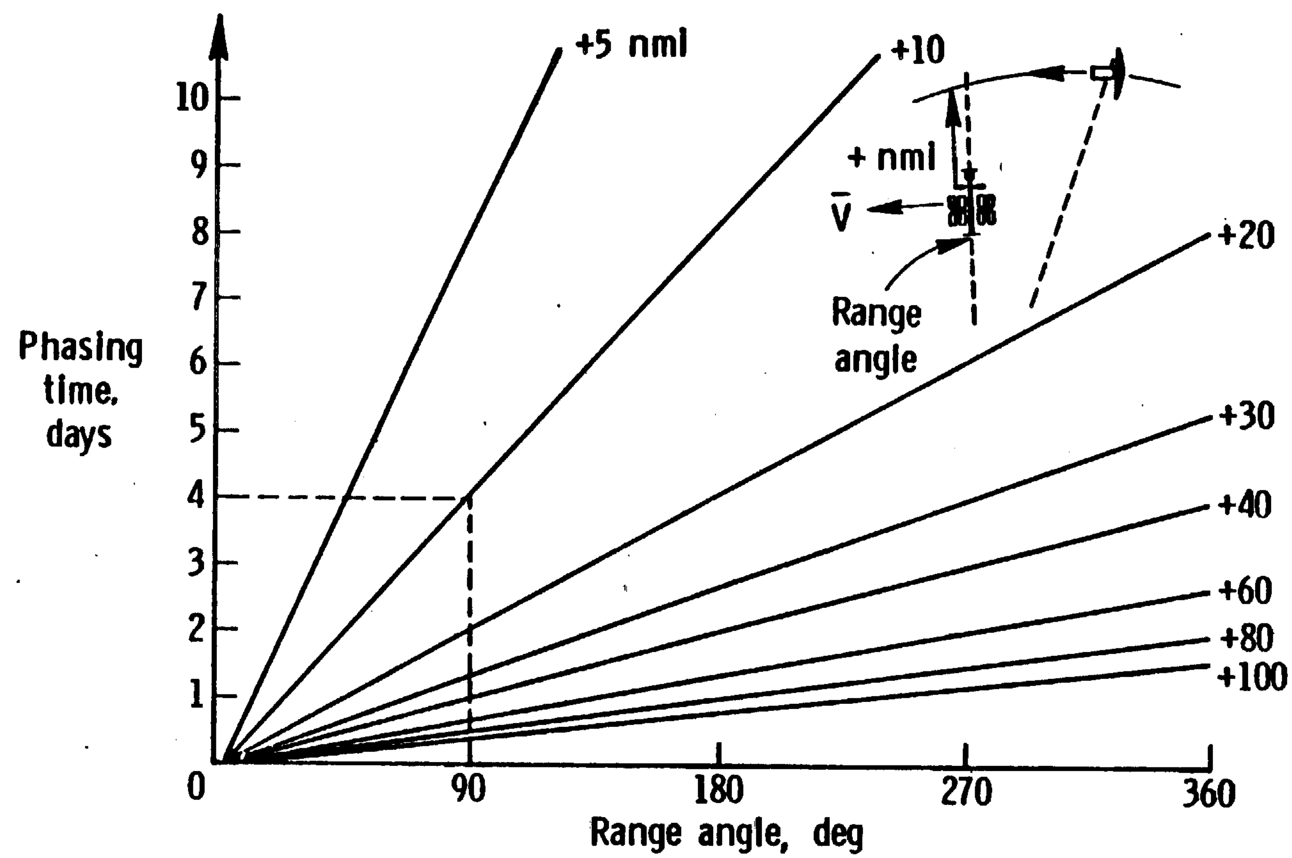


Figure 6. GEO to LEO Transfer (Phasing Time to Reach Hohman Transfer Points)

RELATIVE COORDINATE SYSTEM

The consequences of non-optimum rendezvous maneuvers initiated from positions other than Hohmann transfer points were examined by calculations based on simplified Clohessy-Wiltshire equations derived by Eggleston (NASA TR R87). The relative coordinate system shown on Figure 7 is centered at the Space Station c.g. that is in a circular orbit. The local gravity vector is assumed parallel to the Y axis which is positive up along a radius vector from the center of a geocentric Earth and is a limiting factor in this analysis. The X axis is positive along a negative tangent to the orbital velocity vector, and the Z axis is perpendicular to the orbital plane to complete the right-hand coordinate system. This analysis only considered transfers within the orbital plane. The range of relative altitude for the intermediate parking orbit were within 10 n.mi. above and below the Space Station orbit and within relative range angles that lead or lag the Space Station by 2°. These positions were assumed to be within a zone controlled directly by the Space Station with complete and accurate state vector information about the vehicle.

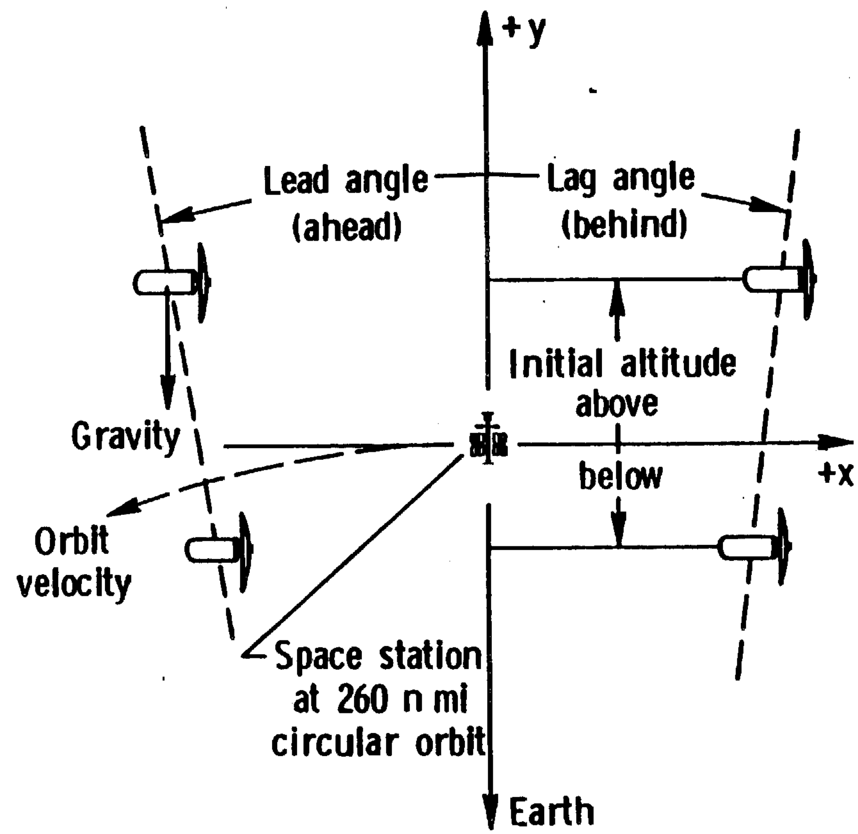


Figure 7. Relative Coordinate System

RENDEZVOUS TIME AND TRANSFER VELOCITY VARIATIONS

Variations in the desired time to arrive in proximity of the Space Station in a 260 n.mi. circular orbit and the corresponding transfer velocity requirements are shown on Figure 8. The initial conditions for these rendezvous trajectories are in each of the four quadrants about the Space Station in the plane of the orbit. The general trends for these variations are approximately the same for all quadrants with minimum transfer velocities ranging from about 10 to 100 ft/sec. The transfer times corresponding to these velocities vary from 50 to 80 minutes with longer transfer times occurring for the larger initial range angles. Significantly increasing velocity penalities result when the desired transfer times are reduced either below 30 minutes for the larger initial range angles or increased beyond 90 minutes for most initial conditions. The minimum variations in transfer velocity occur when the vehicle is directly above or below the Space Station, which are close to the Hohmann transfer points. Neither the AOTV nor the OTV design is significantly impacted by an increase of 100 ft/sec in velocity requirements, nor does a rendezvous transfer time of 50 to 80 minutes appear to be critical at this time. Therefore, other discriminators were sought.

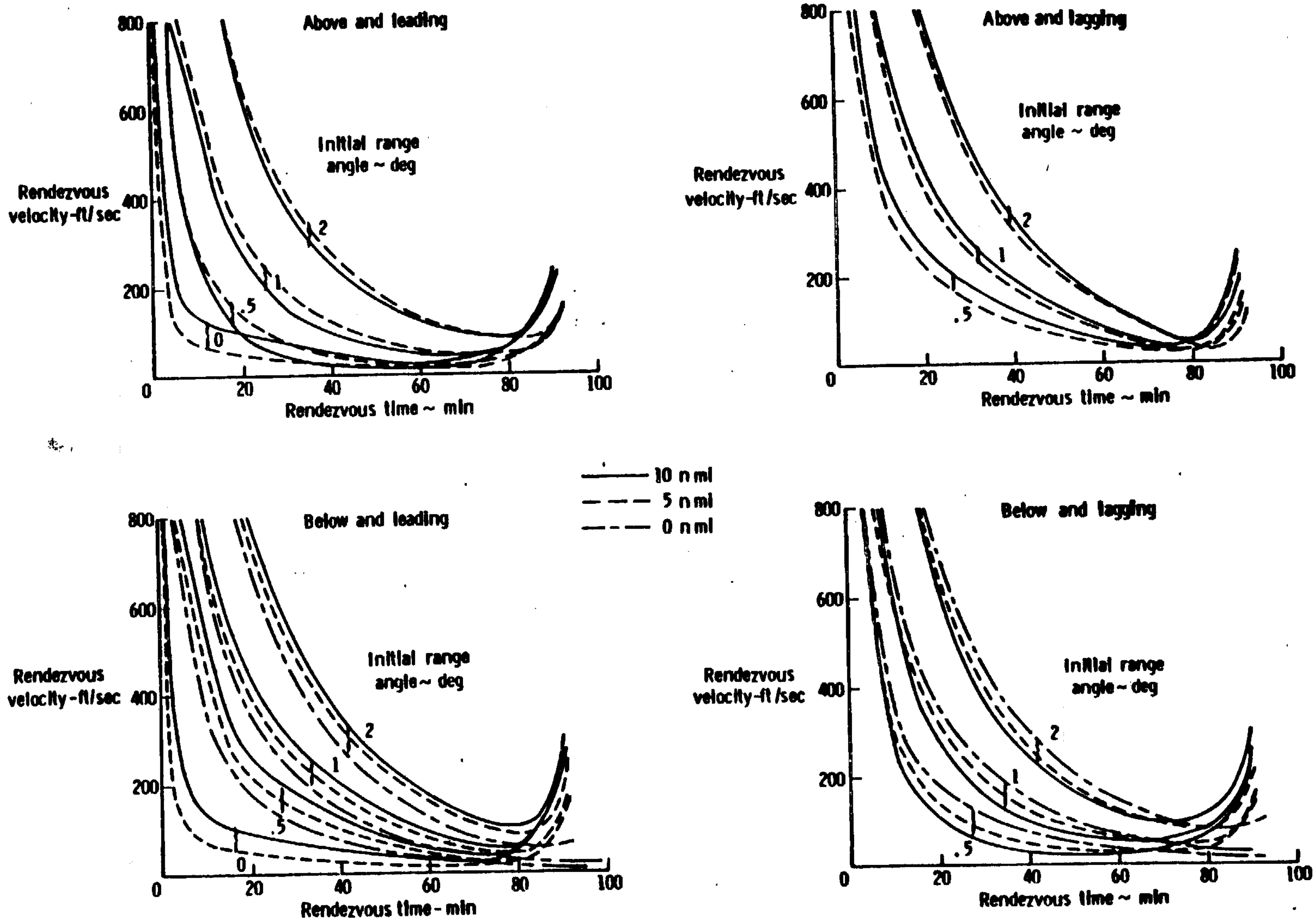


Figure 8. Rendezvous Time and Transfer Velocity Variations

TERMINAL CONDITION VARIATIONS

The magnitudes of final approach velocities and flight-path angle at an arbitrary 500 ft from a Space Station in a 260 n.mi. circular orbit are shown on Figure 9. These conditions result from selected minimum Δv planar rendezvous transfers shown previously on Figure 8. The near tangential approaches to the Space Station (0° and 180° flight-path angles) result from rendezvous transfers initiated at positions either above and ahead or below and behind the station. If transfer was initiated from a co-orbiting position behind the station (0.0 n.mi.), a tangential approach would result. The opposite initial conditions produce steep flight path angle approaches to the station.

The magnitude of the arrival velocities are indications of the efficiency of the transfer maneuver because they represent the approximate residual energy that must be dissipated before berthing and docking can proceed. The larger arrival velocities also have the potential for more contamination of the Space Station because of longer and/or higher levels of thrust. The smallest relative velocities upon arrival result from initial positions above and ahead or below and behind with little difference noted when initiated ahead or behind on a co-orbiting altitude.

These results indicate that to minimize potential contamination, the perferred approaches to the station are initiated from zones above and ahead or below and behind the Space Station. When initiated from co-orbiting positions, a preferred zone would be from behind the station. The resulting tangential approaches provide more efficient capability for co-orbiting (V bar) station keeping and create less concern about the relative position of station superstructure. There is still a concern about a safe waveoff. However, the transport vehicle and station architectures could change what may be considered as a safe approach and requires a full 6 DOF analysis with sensors and guidance systems included.

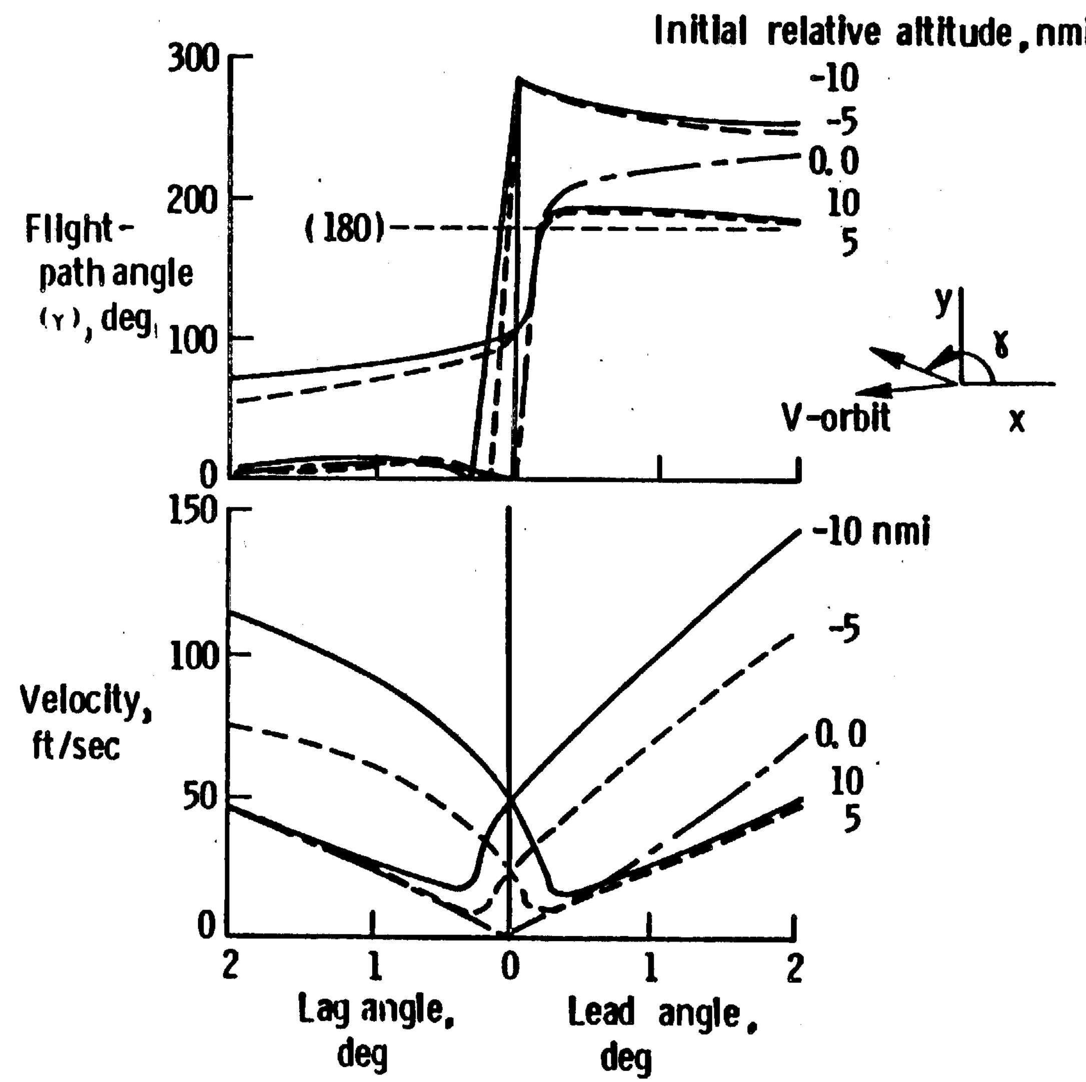


Figure 9. Terminal Condition Variations

RELATIVE APPROACHES TO THE STATION

The trajectories of an AOTV to the Space Station (Figure 10) from initial positions on a coplanar parking orbit either 10 n.mi. above or below and 2° ahead or behind the station illustrate some concerns in designing safe approaches to the station. The planar motion relative to the Space Station that is plotted on the left portion of the figure for approaches from ahead and above or below and behind the station result in the smallest variation in total angle required by the Space Station for tracking the AOTV to within proximity of the station and, therefore, in reduced angular motion for the tracking antenna.

Communications with an AOTV of this particular design could be limited if the material selected for the thermal protection system on the brake proves to be opaque to the signal frequency. Special timed maneuvers, before and during transit, designed to remove the thermal shield from the line of sight of the communications link may be required to achieve a successful berthing and docking.

The illustrated final approach reveals other concerns and problems. The tangential approach to a target at the station requires lower braking velocities to achieve a station keeping position but still requires proper clearance between the two elements of the space system should unsafe conditons require a waveoff. This is especially true in the case of an AOTV with a potentially large diameter heat shield. The available field of view for tracking may also cause a problem without adequate information about vehicle attitude.

Retrothrusts in vicinity of the Space Station can cause contamination and damage from plume impingement on sensitive station elements. These outcomes depend on the propellant used, allowable levels and types of contaminants, and the actual plume shape. Although the severity of these outcomes is unknown, they can impact the AOTV design in terms of type of thrusters and their locations, thrusting-time, size, and the minimum allowable relative distance to start thrusting.

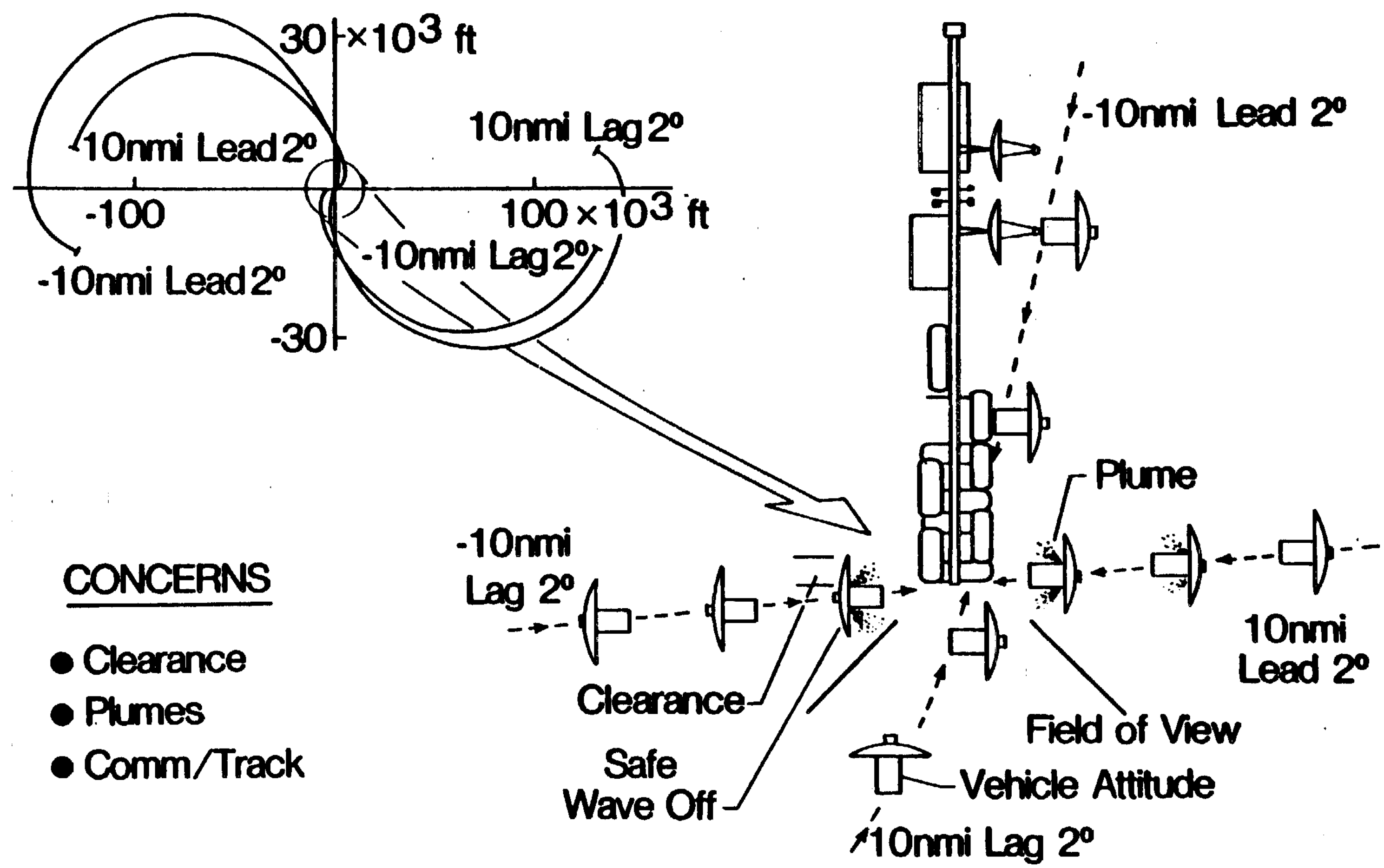


Figure 10. Relative Approaches To The Station

BERTHING AND DOCKING MECHANISMS

Potential solutions to the problems of contamination and collision avoidance include: (1) dual fuel maneuvering systems on the transport vehicle with one non-contaminating fuel; (2) retrieval and placement of the transport vehicle by an auxiliary vehicle that uses a non-contaminating fuel such as a cold gas; and/or (3) use of an extendable berthing and docking mechanism that provides a berthing target away from the station, grapples the transport vehicle, and, with damping and momentum transfer to the station, subsequently retracts and docks the vehicle to the appropriate location on the station. The first two options are essentially dual-fuel systems that address the contamination problem at an expense to the transportation system in terms of low Isp and additional structure. The burden of the auxiliary vehicle falls on the station in terms of additional maintenance, tracking time, and rendezvous planning. The issue of allowable contaminants and/or levels of contamination is not resolved. Neither of the first two options reduces the problem of clearance and collision avoidance. The third option, illustrated on Figure 11, is one of many possible mechanisms that have the potential for reducing contamination levels, regardless of type, and collision problems. These mechanisms (berthing beams) require intelligent end effectors for grappling, capability for out-of-plane motion, insensitivity to lighting conditions, and reliable actuators. The flexible transfer tunnel that is shown is based on the serpentine structure that has been proposed by Mikulas at LaRC and can also be used as a grappling fixture if reliability is demonstrated. All these systems need to be analyzed for reliability, rigidity, response time to compensate for off-nominal arrival conditions, and compatibility with the station control system.

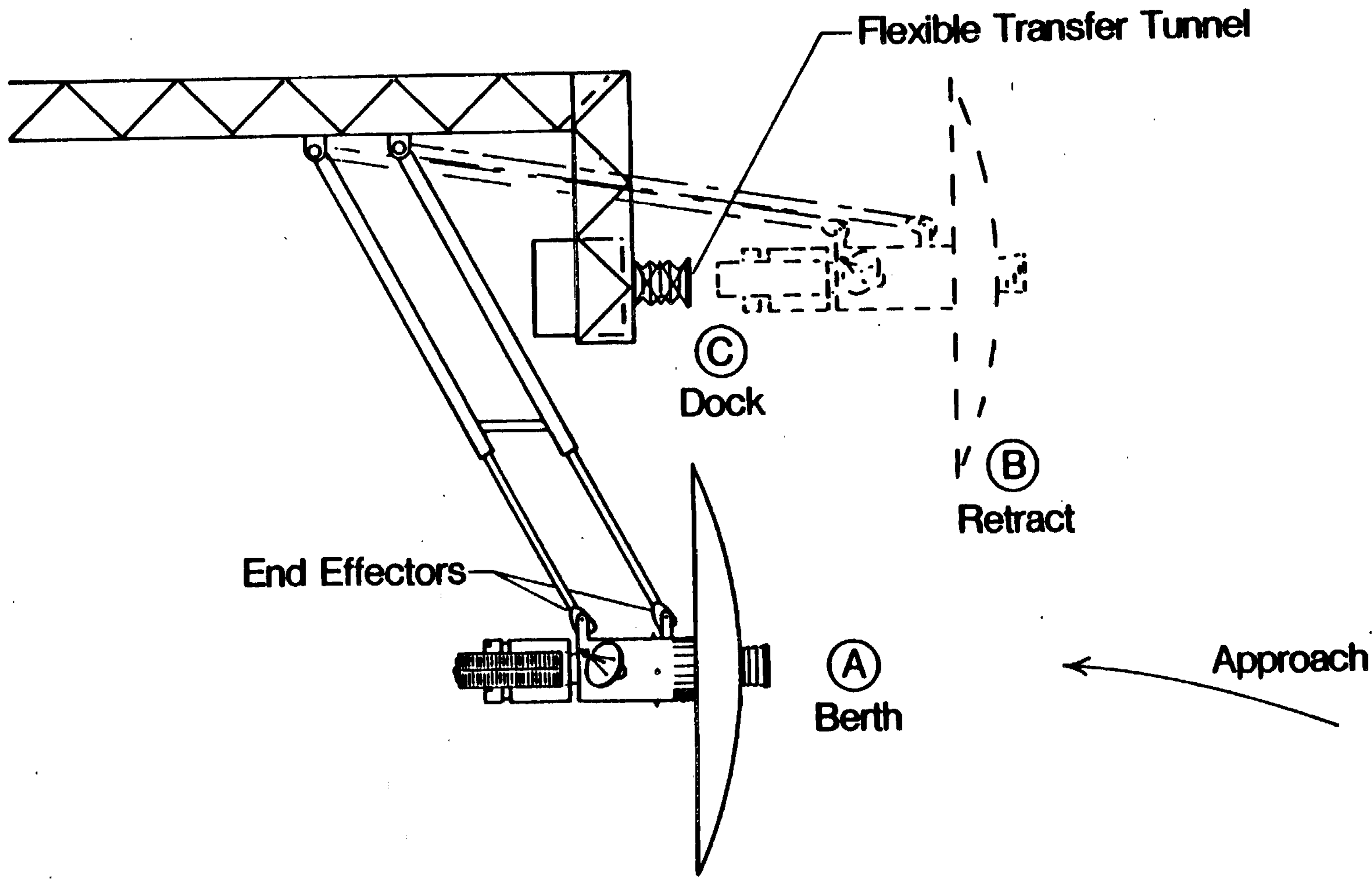


Figure 11. Berthing and Docking Mechanisms

PASSIVE RADIAL (RBAR) SEPARATION

The berthing and docking mechanism (berthing beam) can also aid in the separation of the transport vehicle from the Space Station without significant contamination or plume impingement. The procedure for this separation is essentially the reverse of a docking maneuver along the Earth radius vector (Rbar) but without any thrusting in proximity of the station. In the Rbar separation procedure, the transport vehicle is initially placed at some distance below the Space Station along the Earth radius vector by extending the berthing beam. The Space Station then stabilizes the system orbit that may have been changed with the changing c.g. Thus, the inertial velocity of the transport vehicle is the same as the station. The transport vehicle is then released and allowed to drift as illustrated on Figure 12. At the time of release, the orbit of the vehicle is not the same as the station because of a c.g. shift. The lower altitude of the vehicle c.g. is now the apogee of the new orbit. The drift, because of orbital mechanics, continues for half an orbit, placing the vehicle at perigee and a significant distance away from the station for safer thruster firing. Figure 12 shows the relative positions produced by orbital mechanics and passive separations from positions at the bottom of the station (about 80 feet below the combined c.g.) and 250 feet below the station bottom. The AOTV drifts to a distance of about a half mile to over one mile from initial positions at the bottom or 250 feet below the station, respectively. These distances are reached in 47 minutes or one half the Space Station orbit period. From this point, low thrust firings would be used to reach the safe distance required for main engine ignition. Again, the compatibility of these systems with the station needs to be analyzed.

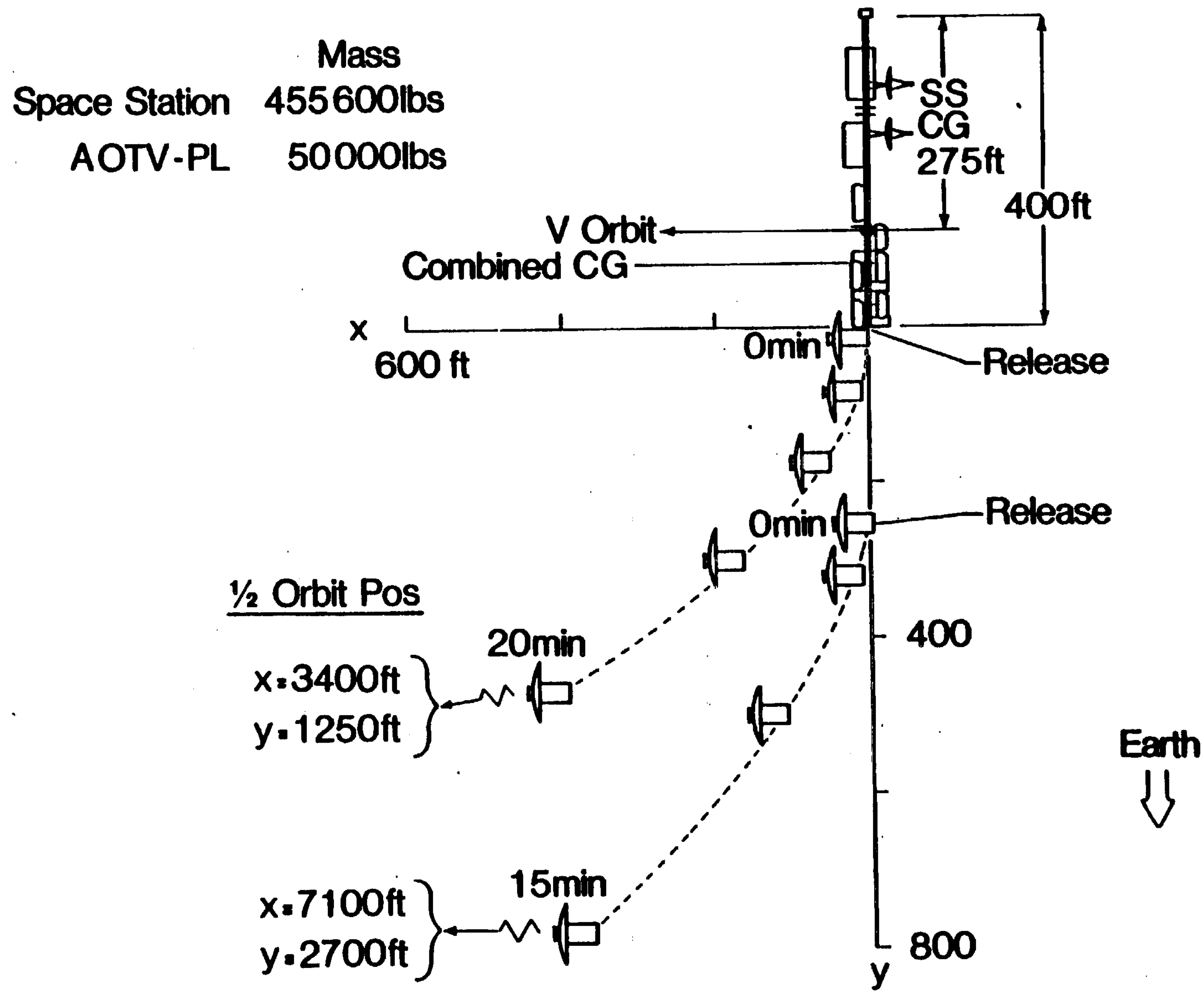


Figure 12. Passive R Bar Separation

ANALYSIS OF PROCEDURES

The demands placed upon the Space Station for maintenance and processing require an analysis of procedures with an analytical tool that is not generally in use for aerospace problems. Different procedures designed to relieve demands on the Space Station for additional time and resources can create a complex system of interrelated events that cannot be evaluated by simple means. A discrete event simulation language can provide a basis for these evaluations. One such language being used at Langley is called SLAM (Simulation Language for Alternate Modeling). Results from analyses with this type of simulation can include estimates of time and resources, identification of potential bottlenecks, and the determination of the level of activity that creates a requirement for additional facilities to maintain an orderly flow through the system. Comparison of results from different simulated procedures provides an excellent basis for evaluation but also requires reliable data from other sources for credible results.

The sample models represented on Figure 13 represent options for different procedures for receiving and processing a space based orbit transport vehicle at the Space Station. Input timelines required for simulating options for berthing and docking include transit and phasing times for OMV utilization, station-keeping time required for grappling, and maneuvering time requirements for direct docking. Also required are the time and resources for OMV maintenance and checkout (C.O.). Similar inputs would be required to evaluate the fueling options. The cargo manifesting options require time and manpower inputs. Locations of facilities on or about the station affect maintenance and processing time because of the time required to safely maneuver key elements of the system on and about the station. Results from analyses of maintenance and processing procedures can impact an orbit transfer vehicle concept design and/or material selection and create demands for additional support equipment. These results cannot be extrapolated directly from experiences with ground procedures. For example, evaluation of an AOTV concept should include the maintenance requirements of the TPS at the station after each flight, which, because of limited manpower, would be accomplished by one person (EVA) with another as a standby. Current estimates (telecon KSC) of the inspection time required for ground inspection of the orbiter is about 0.045 m hrs per ft^2 of TPS area. If this inspection time is required for the AOTV, it would take one person about 16 days at 8 hours a day to visually inspect the 60-ft diameter heat shield. This does not include the EVA overhead of prebreathing and access time or recalibration of the communciation link through the TPS as required with the current orbiter when TPS material is replaced. Recalibration assumes the TPS to be relatively transparent to the communication signals. Obviously, a remote NDT is required to perform this inspection, particularly when the number of AOTV flights increase.

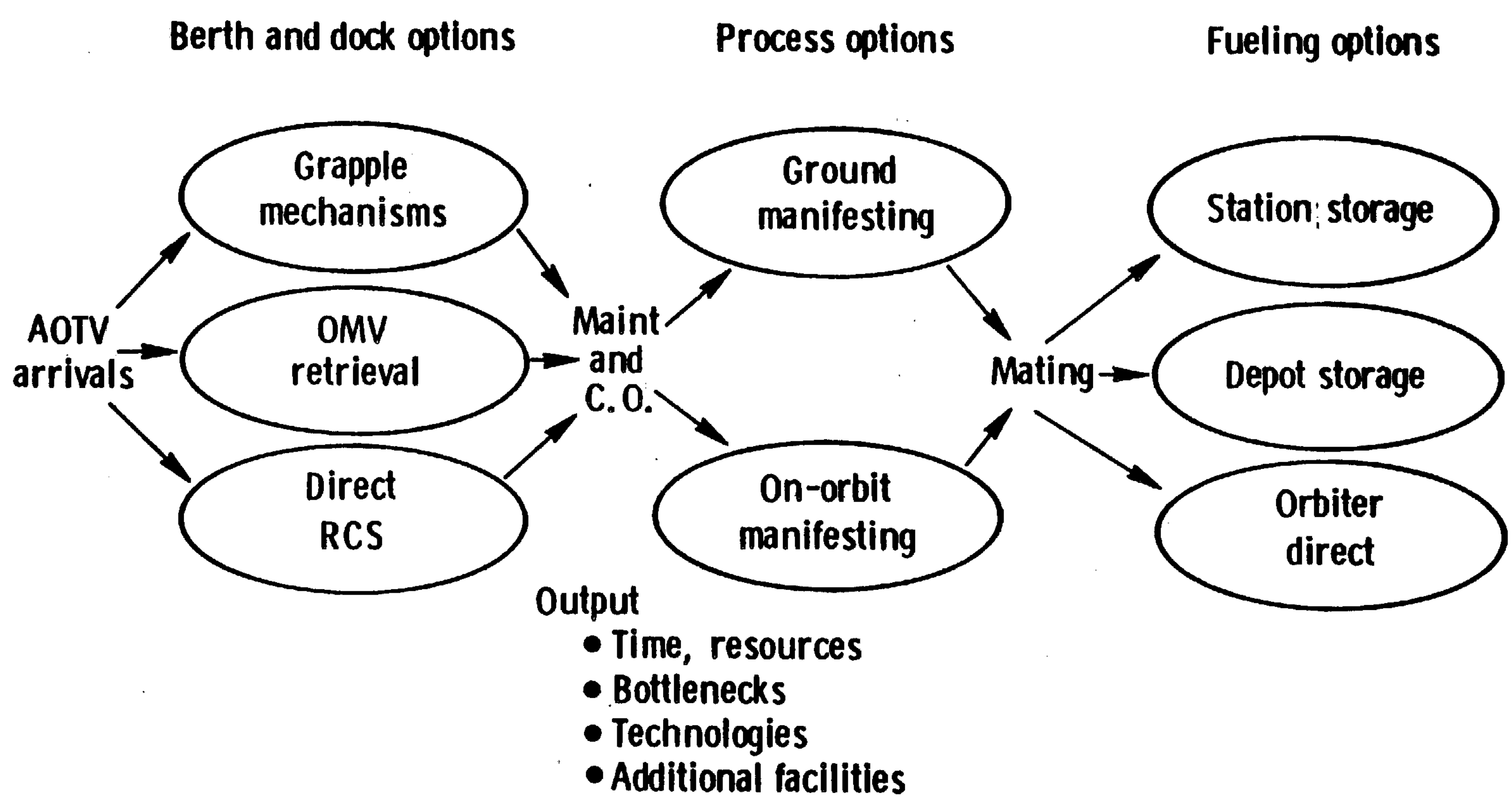


Figure 13. Analysis of Procedures (Discrete Event Simulation Model)

SUMMARY REMARKS

The need to analyze the space transportation system and the Space Station as a complete integrated system to produce overall operational efficiency has been demonstrated. Without a systems approach, many issues, problems, and potential solutions could remain unidentified.

The combined factors of mission model and orbital transfer vehicle design and propellant logistics are major drivers in the determination of an efficient space system. The mission model determines the number of facilities and support required at the Space Station, and the orbital transfer vehicle design determines the size of the hangars and propellant tanks at or on the station. The orbital transfer delivery capability and size could be determined by the majority of payload sizes to be delivered and not necessarily the largest payload. Heavier, less frequent payload deliveries could use alternate delivery systems such as expendables or kick stages that supplement the reusable orbital transfer vehicle. Smaller payloads can be manifested into single large cargo deliveries. The actual match of propellant delivery, vehicle capability, and payload size requires a detailed economic trade.

If the delivery capability of the orbital transfer vehicle is determined by the maximum amount of propellant that could be delivered directly to the Space Station by the orbiter (present or future) for direct fueling, the decisions relative to propellant storage at or on the Space Station can be delayed. At a future date, when the number and frequency of transfer missions becomes better defined and when the long term behavior of cryogenic propellant in zero-g is known, a more informed decision can be made about propellant logistics.

Careful scheduling and coordination is required between ground and Space Station control to provide timely, flexible, and efficient returns from GEO to the Space Station. Minimum station contamination from vehicle retrothrust, flexible mission planning, and timely approaches to the station can be achieved from parking orbits either above and ahead or below and behind the station. Actual selection of safe approaches to the station requires a 6 DOF analysis that includes the Space Station, orbital transfer vehicle architecture, and a guidance, navigation, and control system.

A berthing beam that provides a target for an approaching vehicle away from the station for capture by grappling can provide sufficient clearance for a waveoff, minimize station contamination, and can also be used for passive vehicle deployment. Actual allowable station contamination levels and station control requirements can change the viability of a berthing beam.

Different procedures, designed to minimize demands for time and limited resources to perform vehicle maintenance and processing, need to be analyzed for the space environment and evaluated with reliable, credible data and a simulation tool such as SLAM (Simulation Language for Alternate Modeling). This type of analysis can identify bottlenecks, establish a need for new and/or enhanced technologies, and determine the number of arrivals and departures at the station that create the need for additional facilities.

- DEMONSTRATED BENEFIT OF SYSTEM APPROACH AND IDENTIFIED ISSUES AND PROBLEMS
- STRONG INFLUENCE BETWEEN MISSIONS, DESIGNS, AND LOGISTICS
- IDENTIFIED ALTERNATIVE TO LARGE ON-ORBIT PROPELLANT STORAGE FOR EARLY SPACE STATION. MANY IMPORTANT VARIABLES UNKNOWN AND CAN CHANGE OUTCOME
- VEHICLE TRANSFER TO SPACE STATION REQUIRES COORDINATION--GROUND AND SPACE STATION
  - REDUCED TRACKING TIME
  - RETAIN FLEXIBILITY FOR PLANNING
- ESTABLISHED ZONES FOR INITIAL RENDEZVOUS MANEUVER POSITIONS
  - AHEAD AND ABOVE/BELOW AND BEHIND
  - TIMELY, EFFICIENT, AND FLEXIBLE
  - STILL REQUIRES FULL 6 DOF W/GN&C
- EXTENDABLE BERTHING/SEPARATION MECHANISM DESIRABLE
  - AVOID COLLISION AND CONTAMINATION
  - SAFE WAVEOFF
- EVALUATE PROCEDURES WITH SIMULATION MODELING (DISCRETE EVENT)
  - TIME RESOURCES
  - BOTTLENECKS, TECHNOLOGIES, ADDITIONAL FACILITIES
  - REQUIRES DIVERSE, RELIABLE DATA

Figure 14. Summary Remarks

ANALYTICAL TOOLS

The diverse major analytical programs used in this study are briefly described and are an indication of the scope required for this type of analysis.

POST - (Program to Optimize Simulated Trajectories) is a trajectory optimization program used to obtain trajectory mission times and phasing time requirements for orbit transfers and rendezvous.

AVID - (Aerospace Vehicle Interactive Design) includes an interactive weight and sizing program used to produce parametric space-based orbital-transfer vehicle designs. Sensitivities of vehicle designs to additional maneuvering Δv requirements were also obtained.

RENDOPS - (Rendezvous Operations Simulation) is a trajectory analysis program of the non-thrusting Clohessy-Wiltshire equations with a simplified gravity model which was derived by Eggleston (TR R-87). This program was used to obtain qualitative comparisons of vehicle approaches to the Space Station from within the Space Station control zone.

SLAM - (Simulation Language for Alternative Modeling) is a program language used to analyze a complex series of interrelated discrete events to obtain estimates of resources (vehicles and facilities) required to maintain the orderly operation of a system, to evaluate alternate procedures for the ability to reduce time for vehicle maintenance and processing, and to identify potential bottlenecks and technology improvements which can affect operational efficiency.

ACKNOWLEDGEMENTS

I wish to acknowledge the members of the team whose contributions made this presentation possible. The following is a list of the team members and areas in which they were the major contributors:

Mission Model - J. Rehder

Transfer Vehicle Designs - J. Rehder

Mission Analysis - T. Talay

Propellant Handling - R. Witcofski

Proximity Maneuvers - D. Eide

Mechanical Assists - I. MacConochie

Procedures Analysis and Modeling - W. Morris and N. White

G&C Consultation - H. Stone

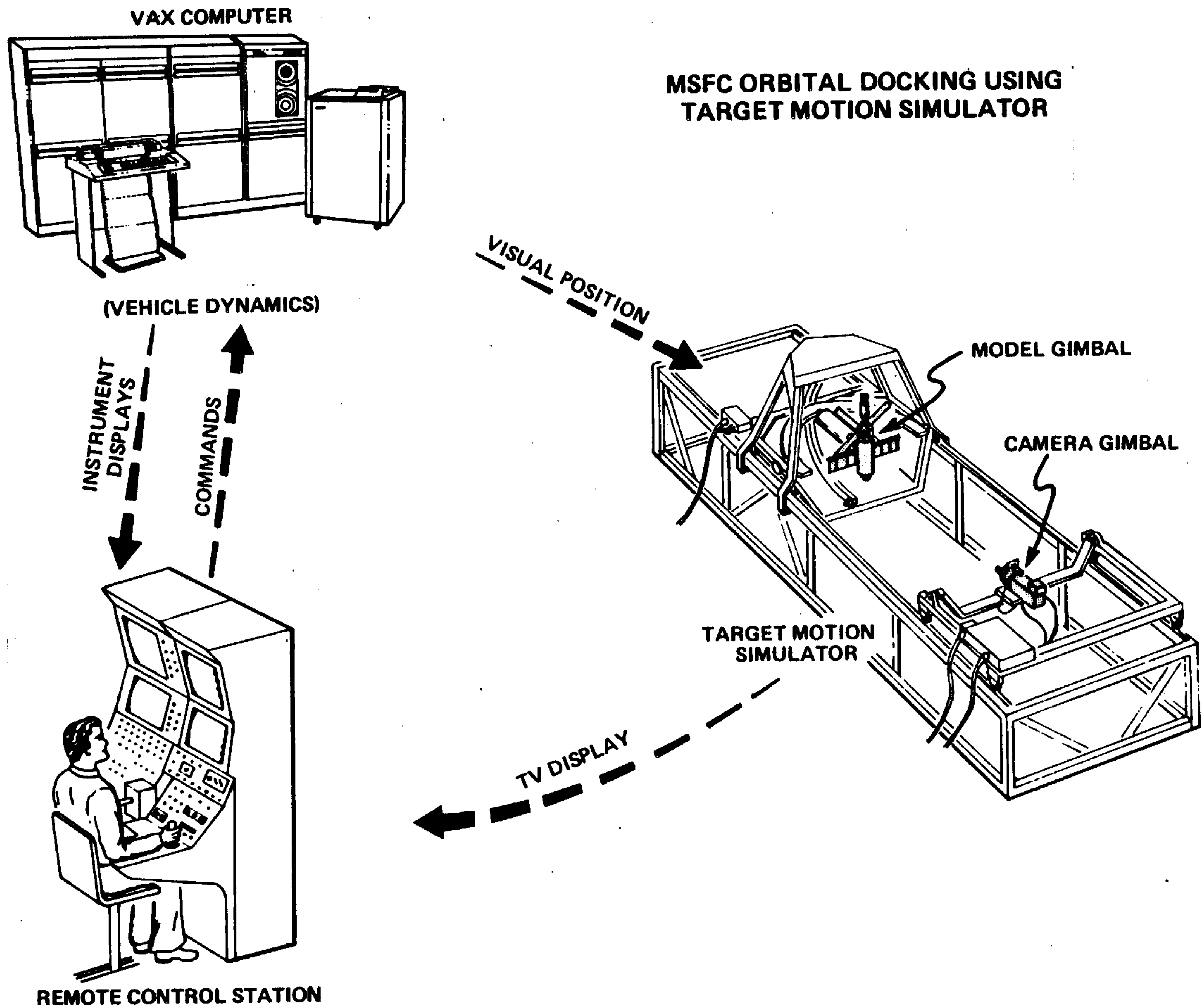
SYSTEM INTEGRATION METHODOLOGIES

FRANK VINZ/MSFC

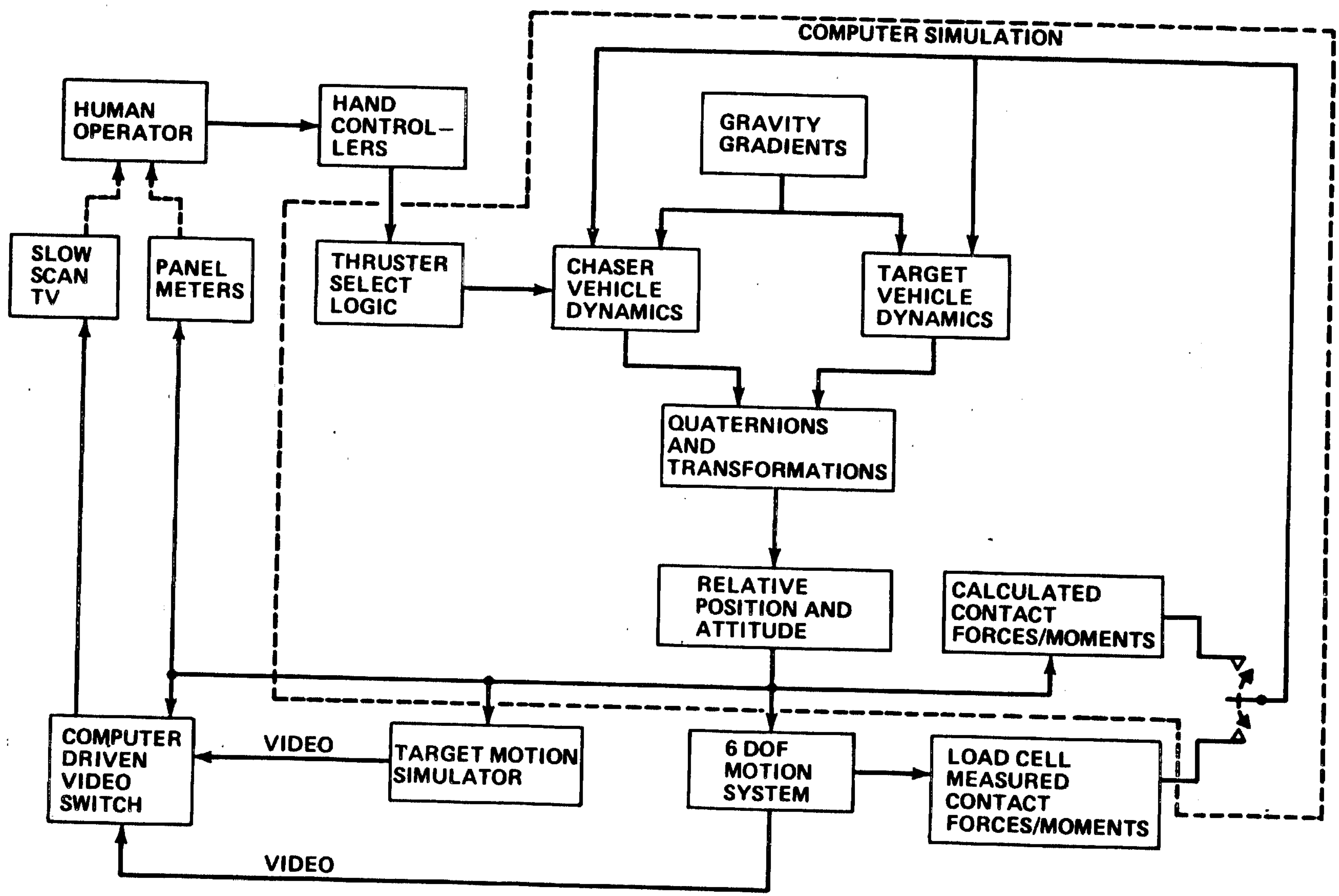
OUTLINE

- OMV DOCKING SIMULATION
- ORBITAL CONTACT DYNAMICS SIMULATION
- SPACE STATION ATTITUDE CONTROL TEST BED
- SPECTRUM OF SYSTEM INTEGRATION APPROACHES
  - SHUTTLE AVIONICS INTEGRATION LAB.
  - IMAGE MOTION COMPENSATION INTEGRATION
  - SSME/HSL
- INTEGRATION METHODOLOGY
  - SYSTEM INTEGRATION GENERALITIES
  - COMPARISON OF INTEGRATION METHODS
  - SELECTION OF INTEGRATION FACILITIES
  - SYSTEM INTEGRATION BY FACILITY EVOLUTION

MSFC ORBITAL DOCKING USING TARGET MOTION SIMULATOR

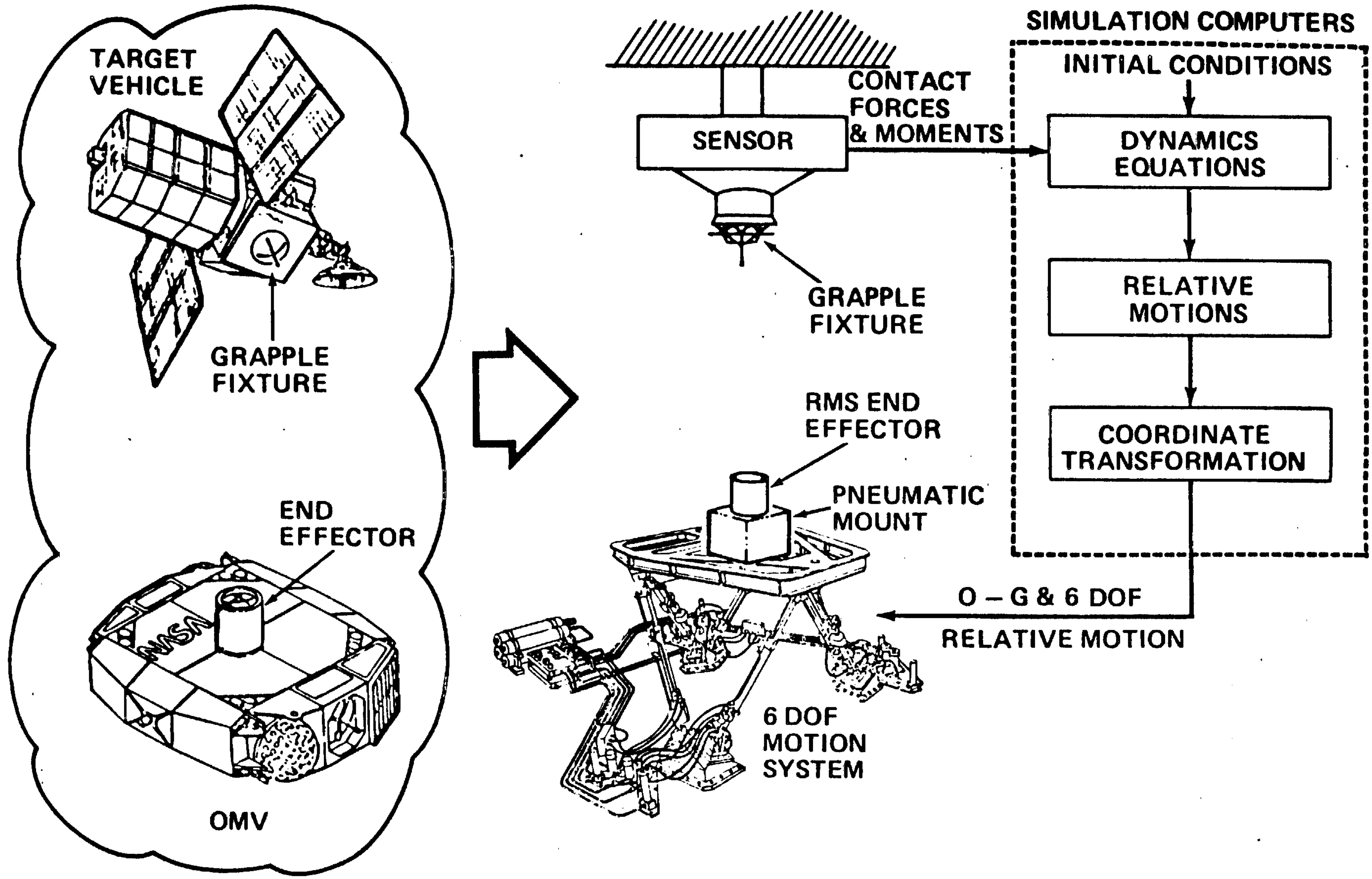


MSFC ORBITAL DOCKING SIMULATION



MSFC MANUAL CONTROL SYSTEM SIMULATION OF OMV

1. FULL SIX DEGREES OF FREEDOM (3 TRANSLATION, 3 ROTATION)
   - SIMULATED RANGE 80 FT. TO DOCK (WITH 10.1 SCALE MODEL OF TARGET)
   - RELATIVE ROTATIONS $\pm 90^o$ PITCH AND YAW; $\pm 180^o$ IN ROLL
2. ORBITAL MECHANICS EFFECT ON RELATIVE MOTION
3. ORBITAL MANEUVERING VEHICLE BASELINE MASS PROPERTIES (MOI, MASS, CG)
4. 24 MANEUVERING THRUSTERS (5#, 10#, 15#) WITH FIRING LOGIC
5. TRANSLATION CONTROLLER (ACCELERATION MODE, ON/OFF)
6. ROTATION (ATTITUDE) CONTROLLER (RATE PROPORTIONAL MODE – CONTROL LOOP CLOSED THROUGH RATE GYROS)
7. TV IMAGE FROM FIXED CAMERA AND SIMULATED RANGE/RANGE RATE MEASUREMENTS PER OMV RADAR MODEL.
8. COMPLEX 3 AXIS MOTION OF TARGET SPACECRAFT
9. RATE HOLD MODE FOR ROTATION CONTROLLER
10. SLOW TV FRAME RATES AND TIME DELAYS
11. VARIABLE OMV MASS PROPERTIES
12. VARIOUS TYPES OF DOCKING TARGET CONFIGURATIONS



CONTACT DYNAMICS USING 6 DOF MOTION SYSTEM

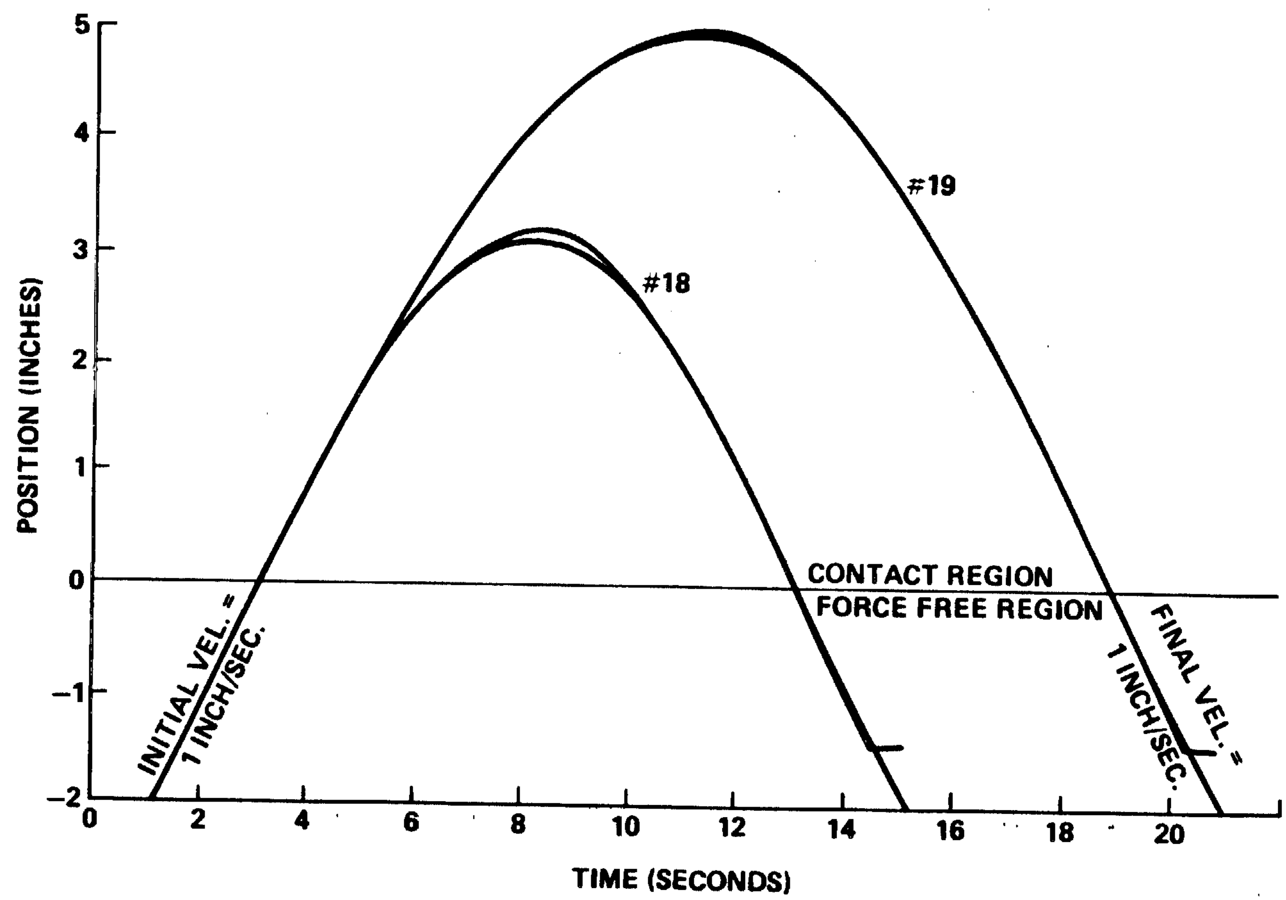
MSFC CONTACT DYNAMICS FOR DOCKING MECHANISM EVALUATION

A. SIX DEGREE–OF–FREEDOM (6 DOF) MOTION SYSTEM FEATURES
   1. HYDRAULICALLY DRIVEN, COMPUTER CONTROLLED
   2. TRANSLATION OF 6–8 FEET
   3. ROTATIONS OF 50–64 DEGREES
   4. PAYLOADS UP TO 20,000 LBS.
   5. ACCELERATIONS OF 2–4 G'S

B. ADAPTATIONS FOR CONTACT DYNAMICS
   1. UNIVERSAL MOUNT FOR ATTACHMENT OF MECHANISM TEST ARTICLES
      (A). PNEUMATIC CONTROL OF MAXIMUM FORCES
      (B). SHEAR PINS FOR BACKUP
   2. FORCE/MOMENT MEASUREMENT OF CONTRACT FORCES IN 6 DOF
      (A). FULL SCALE OF 0.7 LBS. TO FULL SCALE OF 7,000 LBS.
      (B). COMPARABLE RANGE OF FULL SCALE MOMENTS
   3. TEST ARTICLE PROVIDES ITS OWN DYNAMIC CHARACTERISTICS: SPRING RATE, DAMPING, FRICTION, LATCHING, ARTICULATION, DRIVE MOTORS, ETC.

C. COMPUTER CONTROL
   1. DYNAMICS OF TWO BODIES REPRESENTED
      (A). TARGET VEHICLE CAN HAVE GENERAL MOTIONS
      (B). CONTROL SYSTEM CHARACTERISTICS REPRESENTED
   2. PROVISIONS FOR MANUAL OR AUTOMATIC CONTROL
   3. COMPUTER DRIVEN DISPLAYS FOR MANUAL CONTROL
   4. ZERO–GRAVITY DYNAMICS, DISTURBANCES, ETC. PROVIDED BY MATH MODEL

IDEAL VS. ACTUAL PERFORMANCE OF DOCKING DYNAMICS USING 6 DOF MOTION SYSTEM



CONTACT DYNAMICS PROTECTIVE FEATURES

A. COMPUTER CONTROLLED LIMITS

1. SIMULATION HALTED IF FORCES OR MOMENTS ARE EXCEEDED
2. SIMULATION HALTED IF CLOSING VELOCITIES ARE EXCEEDED

B. PNEUMATICALLY CONTROLLED LIMITS

1. SIMULATION HALTED IF UNIVERSAL MOUNT RECEIVES X,Y,Z FORCES GREATER THAN PNEUMATIC THRESHOLD (ADJUSTABLE)

C. BREAKAWAY MOUNTING BOLTS

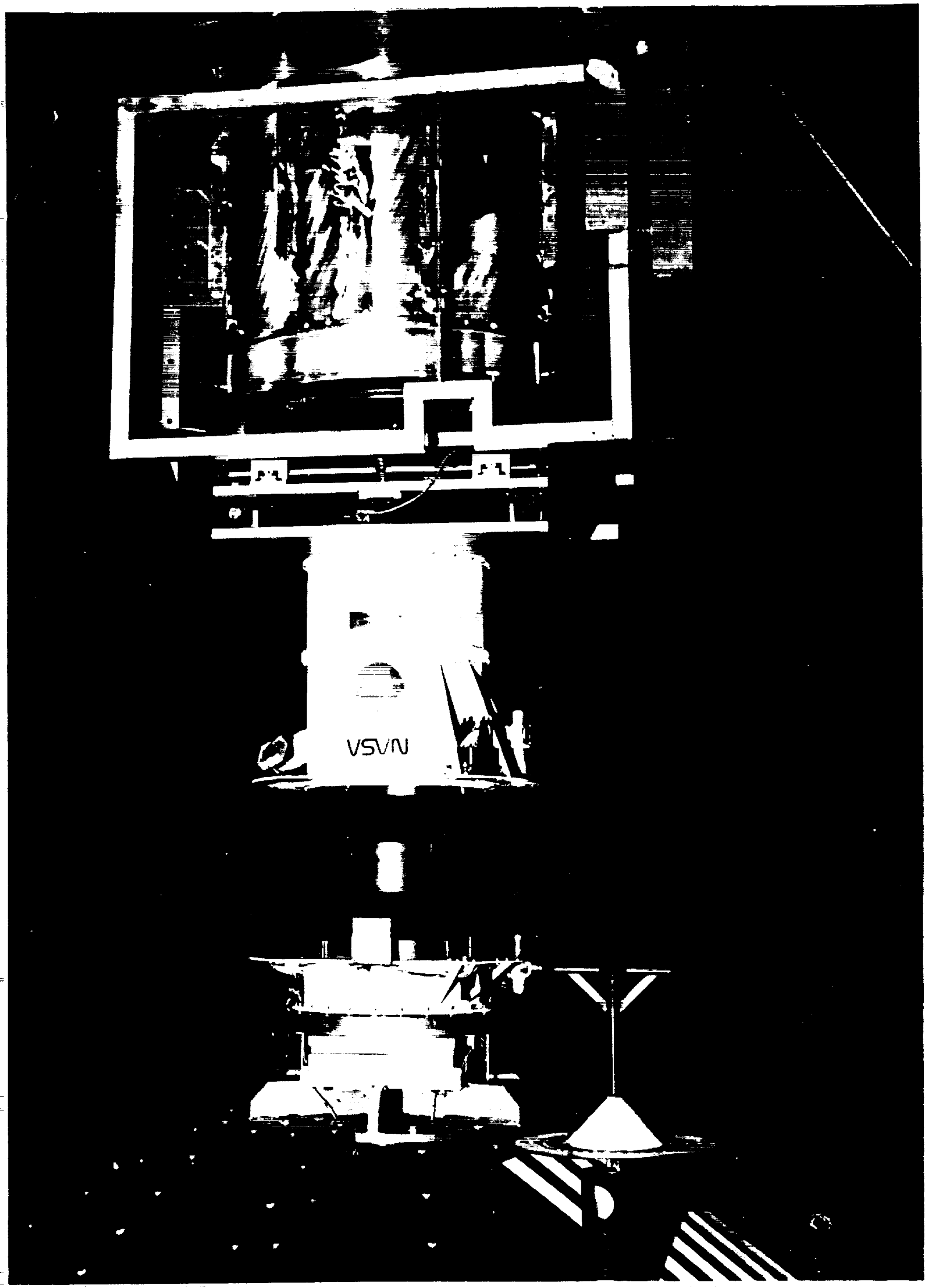
1. SHEAR PIN FOR MOMENT ABOUT VERTICAL AXIS
2. TENSILE LINK FOR VERTICAL FORCE
3. BREAKAWAY BOLTS FOR VERTICAL COMPRESSIVE FORCE AND Y, Z MOMENTS
   - COUNTER WEIGHT/CABLES PROTECT UPPER TEST HARDWARE

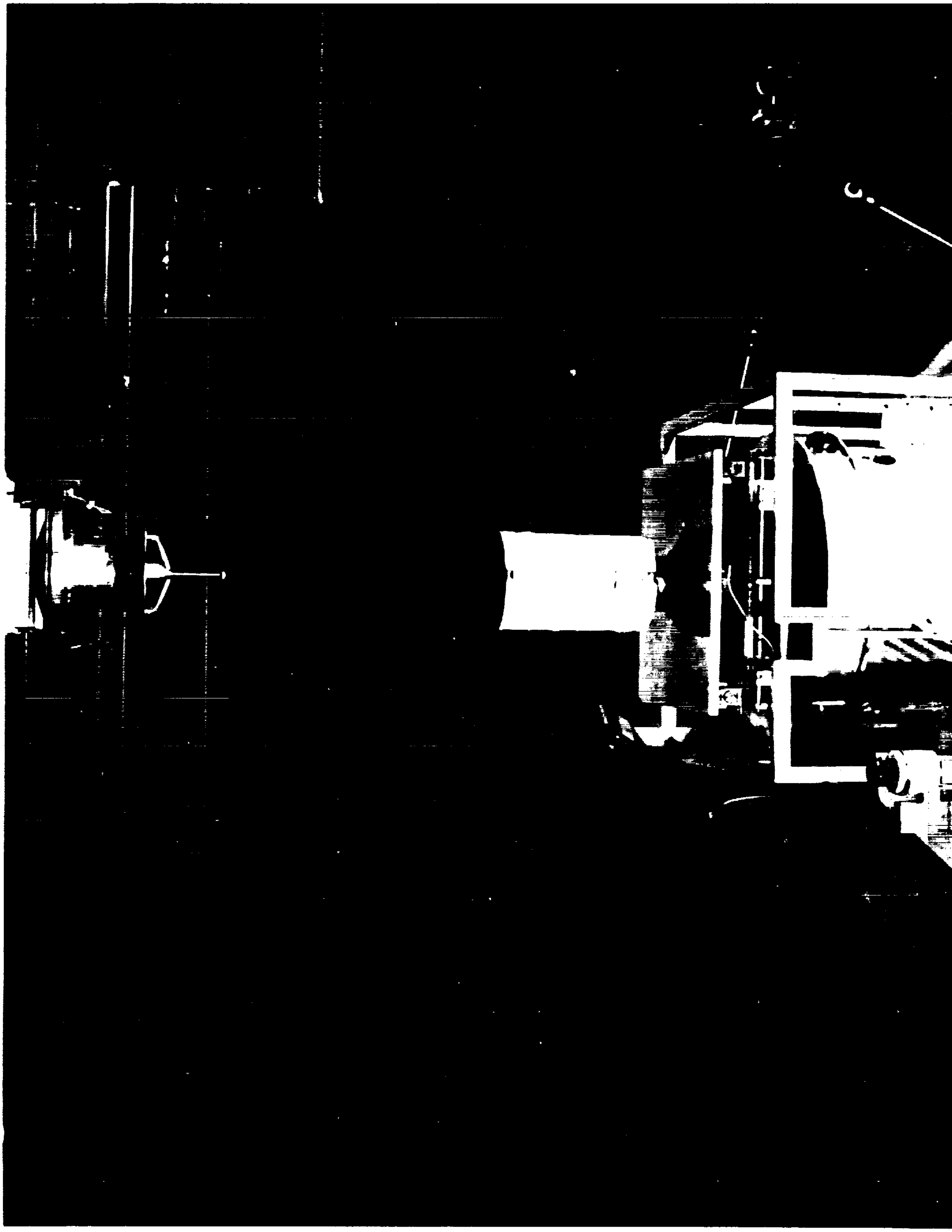
D. 6 DOF MOTION SYSTEM SAFETY FEATURES PROVIDED BY MANUFACTURER

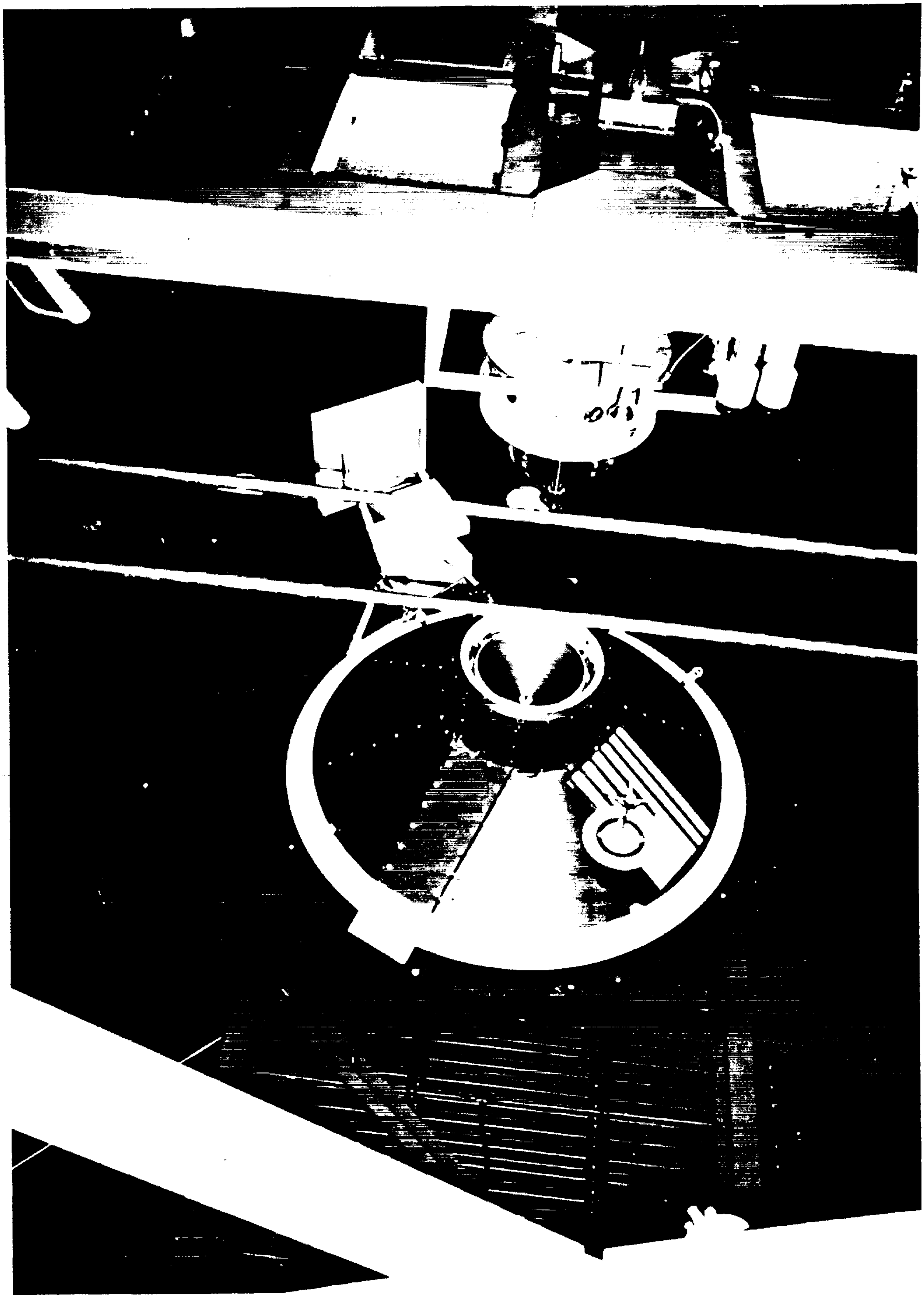
1. HYDRAULIC SHUTDOWN IF POWER SUPPLIES OUT OF RANGE
2. HYDRAULIC SHUTDOWN IF COMPUTER SIGNAL LOST
3. CRUSHIBLE MOUNTS PROTECT HYDRAULIC ACTUATORS

PERFORMANCE OF SIX DEGREE–OF–FREEDOM MOTION SYSTEM

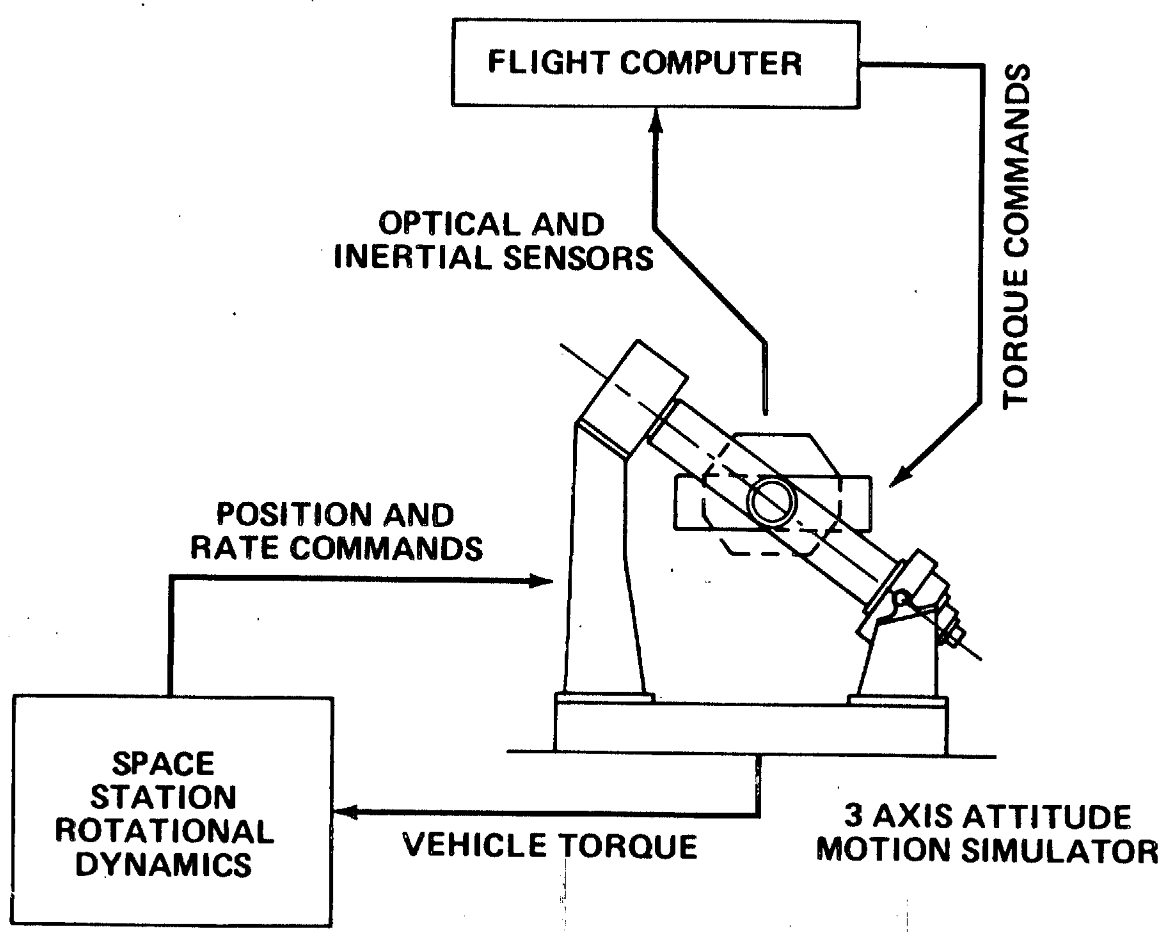
MOTION	POSITION	RATE	ACCELERATION	
			NO LOAD	23K LB/LOAD
PITCH	$+30^{o}$, -20^{o}	$\pm 15^{o}$/SEC	±6.5 RAD/SEC	±2 RAD/SEC
ROLL	$\pm 22^{o}$	$\pm 15^{o}$/SEC	±7 RAD/SEC	±2 RAD/SEC
YAW	$\pm 32^{o}$	$\pm 15^{o}$/SEC	±6 RAD/SEC	±2 RAD/SEC
LATERAL	±48 IN	±24 IN/SEC	±2.4 G	±0.6 G
LONGITUDINAL	±48 IN	±24 IN/SEC	±2 G	±0.6 G
VERTICAL	39 IN. UP	±24 IN/SEC	–2.6 G	–2 G
	39 IN. DOWN		+3.6 G	+3 G







SPACE STATION ATTITUDE CONTROL TEST BED



THE SPACE STATION ATTITUDE CONTROL TEST BED CONSISTS OF:

- A LARGE 3 DEGREE OF FREEDOM TABLE POWERED BY HYDRAULIC ACTUATORS DESIGNED TO GIVE HIGH BANDWIDTH AND EXTREMELY FINE CONTROL (.005 DEG.) THROUGH LARGE ANGLES ($\pm 35^{\circ}$ TO $\pm 180^{\circ}$)
- COMPENSATION FOR EARTH ROTATIONAL RATE
- CONTROL MOMENT GYRO'S
- STAR TRACKER
- RATE GYRO'S
- STAR SIMULATOR AND SOLAR SIMULATOR PROVIDING COLLIMATED LIGHT HAVING THE SPECTRAL CONTENT AND INTENSITY RECEIVED IN EARTH ORBIT

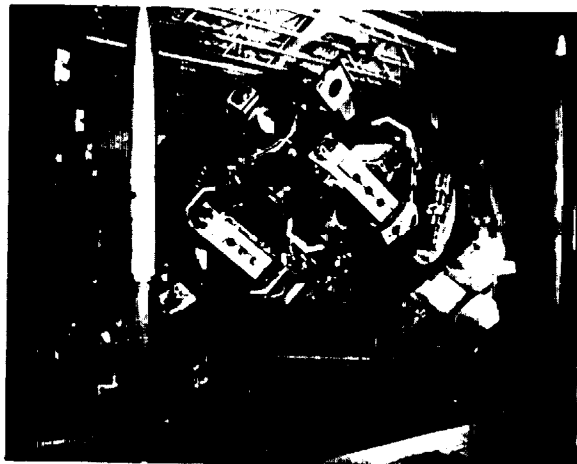
THE CONTROL SYSTEM SIMULATOR INCLUDES A 3 DEGREE OF FREEDOM POINTING MOUNT TABLE

- SEVERAL MISSIONS REQUIRE POINTING MOUNTS
- POINTING MOUNT CONTROL WILL BE HIGHLY INTERACTIVE WITH SPACE STATION CORE CONTROL AND WITH THE DYNAMICS OF THE STRUCTURE

SPACE STATION ATTITUDE CONTROL SYSTEM SIMULATION ACTIVITIES

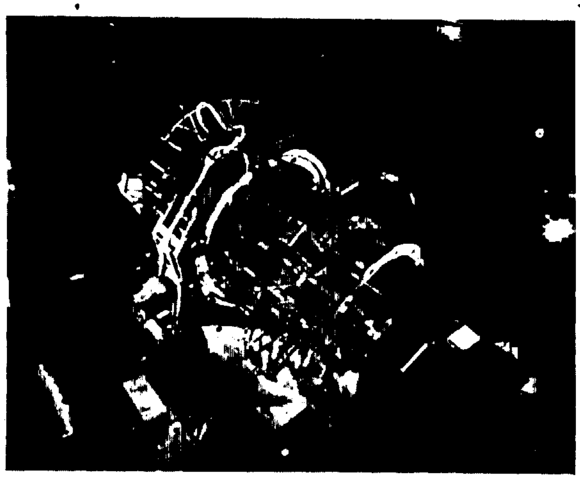
- DEVELOP A REAL TIME SIMULATION OF THE SPACE STATION DYNAMICS AND THE ENVIRONMENT
- EVALUATION OF THE DYNAMIC CHARACTERISTICS OF THE ATTITUDE CONTROL SYSTEM AND NEW MOMENTUM MANAGEMENT CONTROL LAWS
- EVALUATION OF FAULT ISOLATION AND REDUNDANCY MANAGEMENT TECHNIQUES
- EVALUATION OF MODIFIED AND IMPROVED COMPONENTS SUCH AS CMG'S AND RATE GYRO'S
- EVALUATION OF THE TRADE BETWEEN FINE BODY POINTING, FINE POINTING MOUNTS, AND FREE FLYERS
- INTEGRATION AND VERIFICATION OF INTERFACES BETWEEN CONTROL COMPONENTS AND SOFTWARE

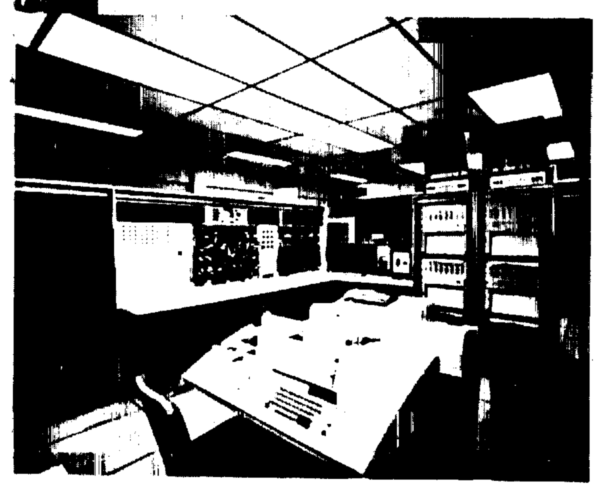
ATTITUDE POINTING SYSTEM SIMULATION LAB


ATTITUDE MOTION SIMULATION

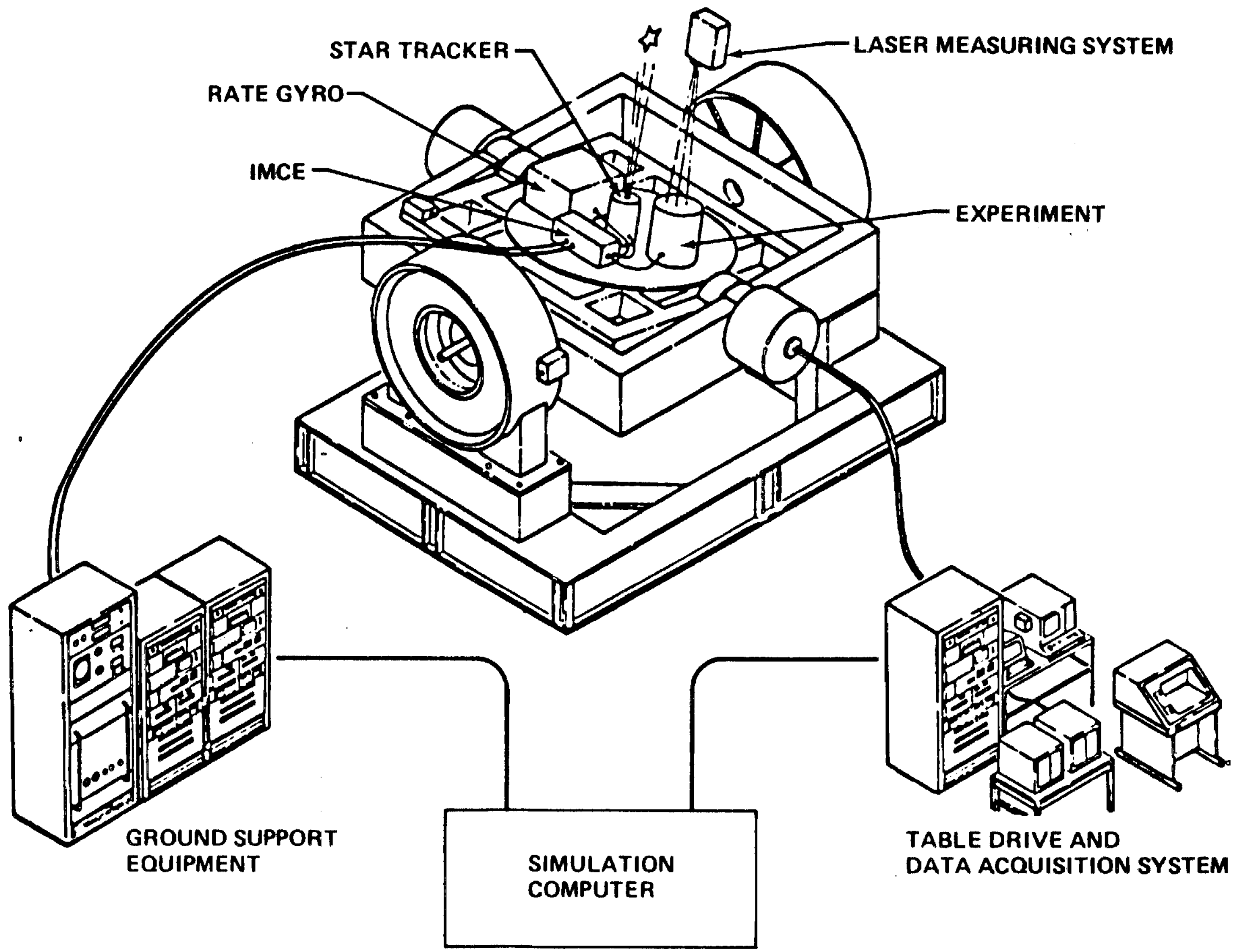

CONTROL CONSOLES


INTERFACE AND CONTROL

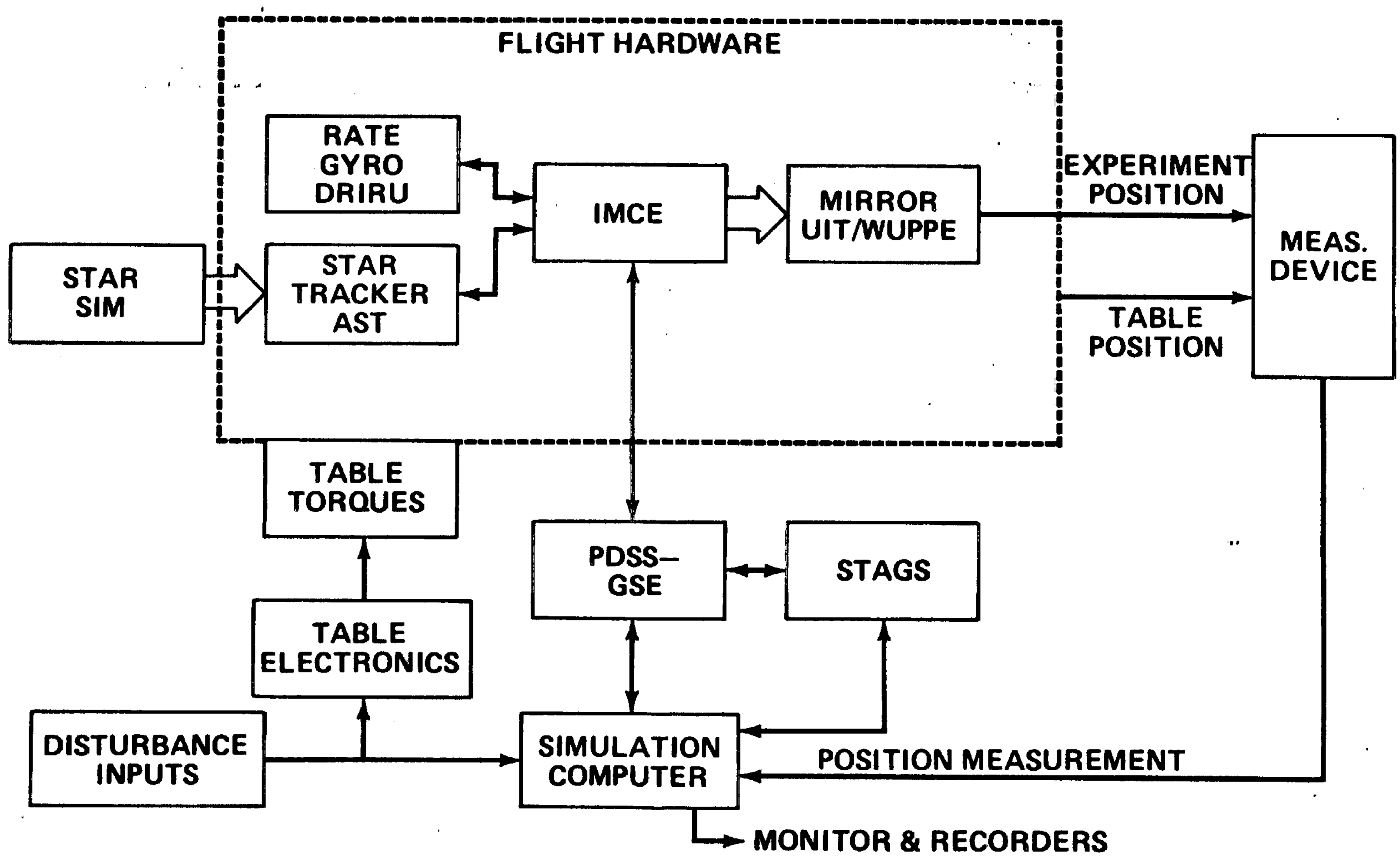

POINTING SYSTEM SIMULATOR


HYBRID COMPUTER

MSFC IMAGE MOTION COMPENSATION SYSTEM INTEGRATION TEST FACILITY (ITF)



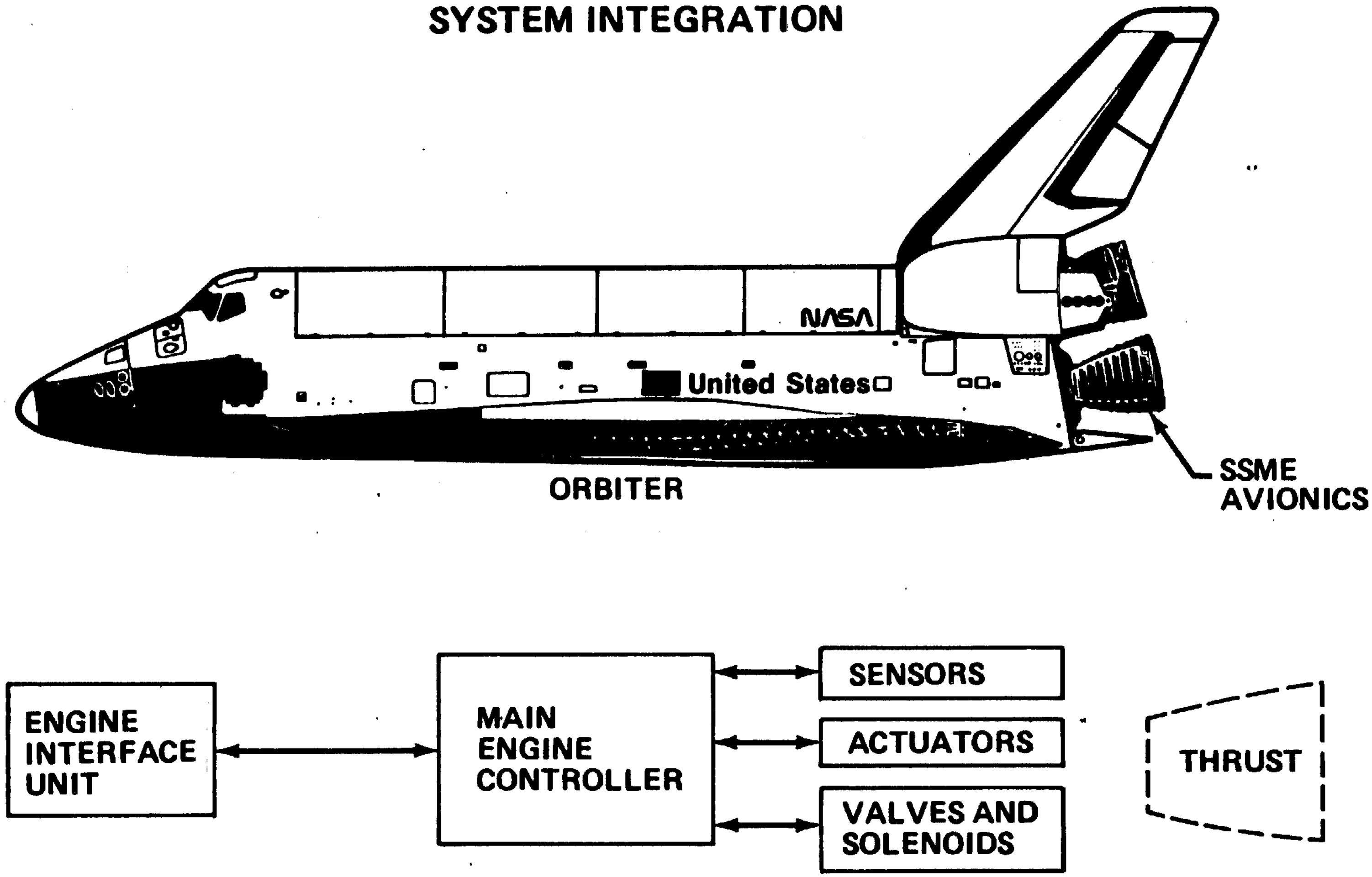
INTEGRATION FACILITY FOR IMAGE MOTION COMPENSATION SYSTEM (ASTRO–1 FLIGHT)



INTEGRATION ANOMALIES DETECTED ON IMAGE MOTION COMPENSATION SYSTEM

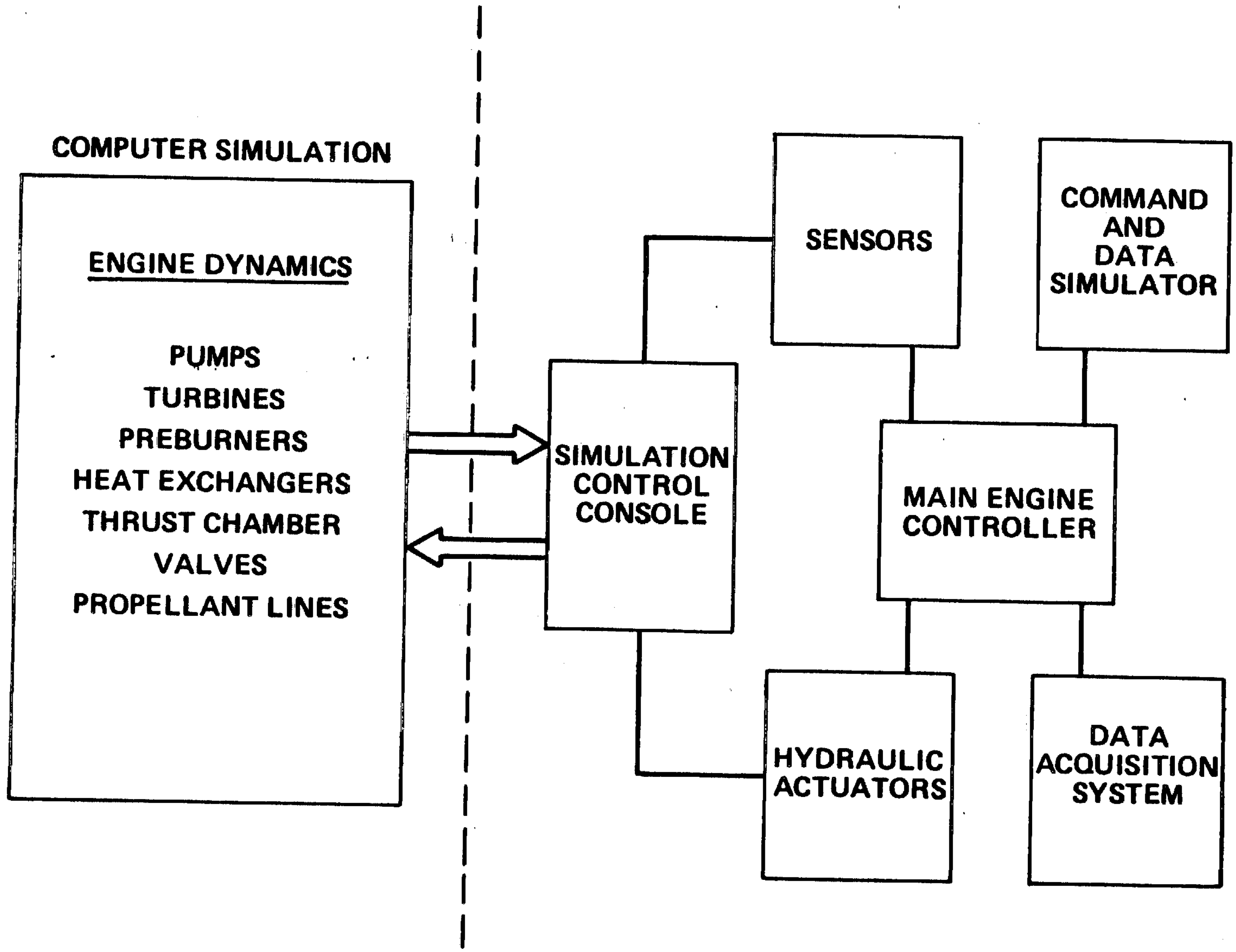
1. EQUATIONS OF MOTION REQUIRED AN ADDITIONAL TRANSFORMATION
2. STAR TRACKER COORDINATE SYSTEM REVERSED
3. CODING ERRORS IN FLIGHT SOFTWARE
4. INCOMPATIBILITY BETWEEN IMCE AND WUPPE SERIAL DATA INTERFACE (REQUIRED CHANGES TO WUPPE)
5. FOUR CABLE/CONNECTOR INTERFACE INCOMPATIBILITIES
6. IMCE OUTPUT DATA CONTAINED INTERMITTENT SPIKES
7. CABLE INTERFERENCE CAUSED DYNAMIC RESPONSE ANOMALY IN UIT MINOR ACTUATOR
8. GIMBAL BEARING PRELOADING ERROR CAUSED NONLINEARITY IN WUPPE MIRROR ACTUATOR.

SPACE SHUTTLE MAIN ENGINE (SSME) SYSTEM INTEGRATION



SSME HARDWARE SIMULATION LAB (HSL)

SPACE SHUTTLE MAIN ENGINE/HARDWARE SIMULATION LAB



SSME/HSL REQUIREMENTS

- INTEGRATION OF FLIGHT HARDWARE AND FLIGHT SOFTWARE
- SOFTWARE VERIFICATION AND HARDWARE VALIDATION FOR EACH FLIGHT AND EACH STATIC FIRING
- EVALUATION OF OFF NOMINAL ENGINE OPERATION
- VERIFICATION OF FAILURE MODES AND REDUNDANCY MANAGEMENT
- VALIDATION OF CONTROL SYSTEM DYNAMICS FOR THE FULL RANGE OF ENGINE OPERATION

SSME/HSL FUNCTIONS

- ADVANCED TECHNOLOGY FOR SPACE SHUTTLE ENGINES
  - DIGITAL COMPUTER FOR CONTROLLER
  - HIGH SPEED TURBOPUMPS
  - LARGE VARIATIONS IN THRUST
- CRITICAL INTERFACES TO OTHER ELEMENTS

ORIGINAL PAGE

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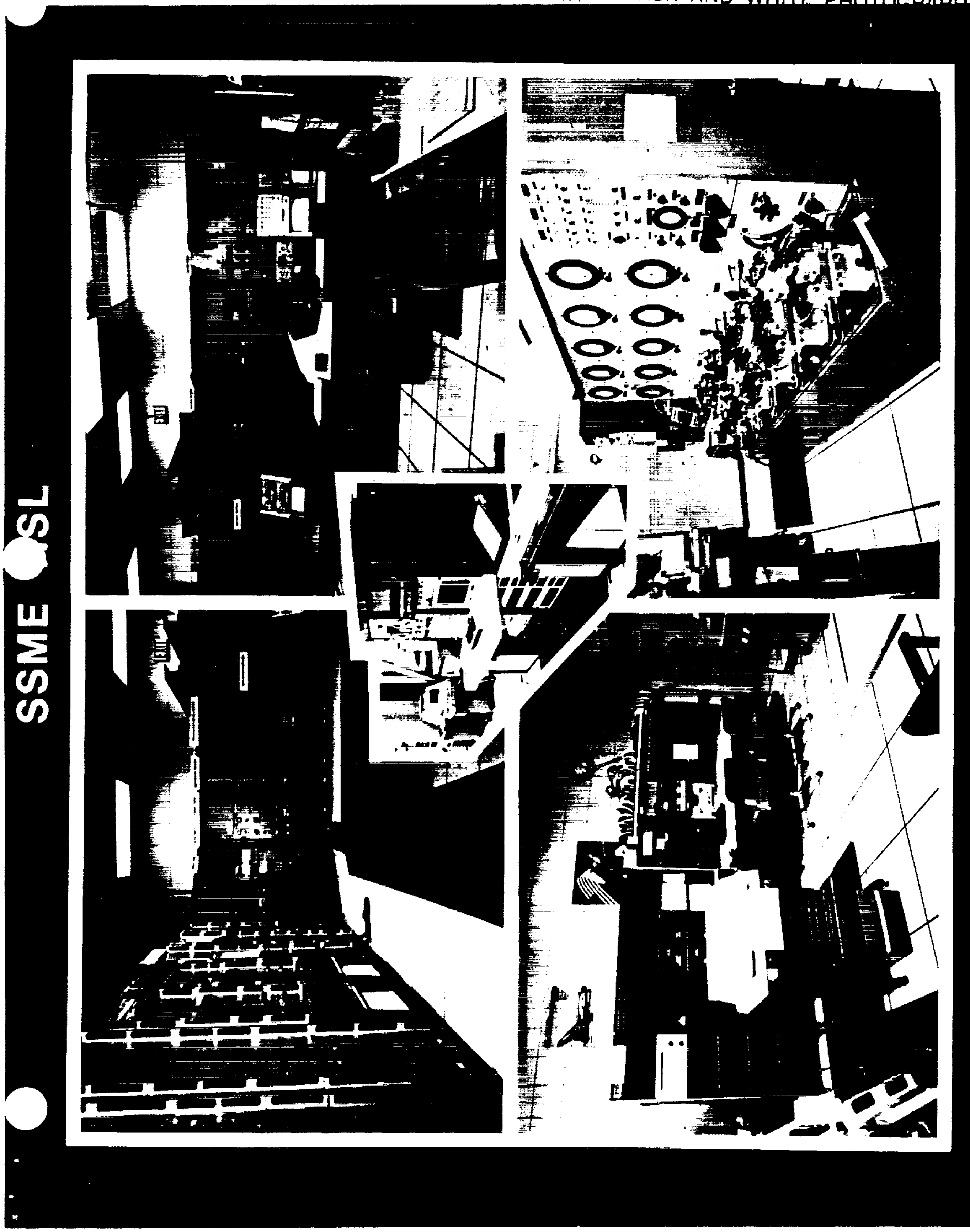
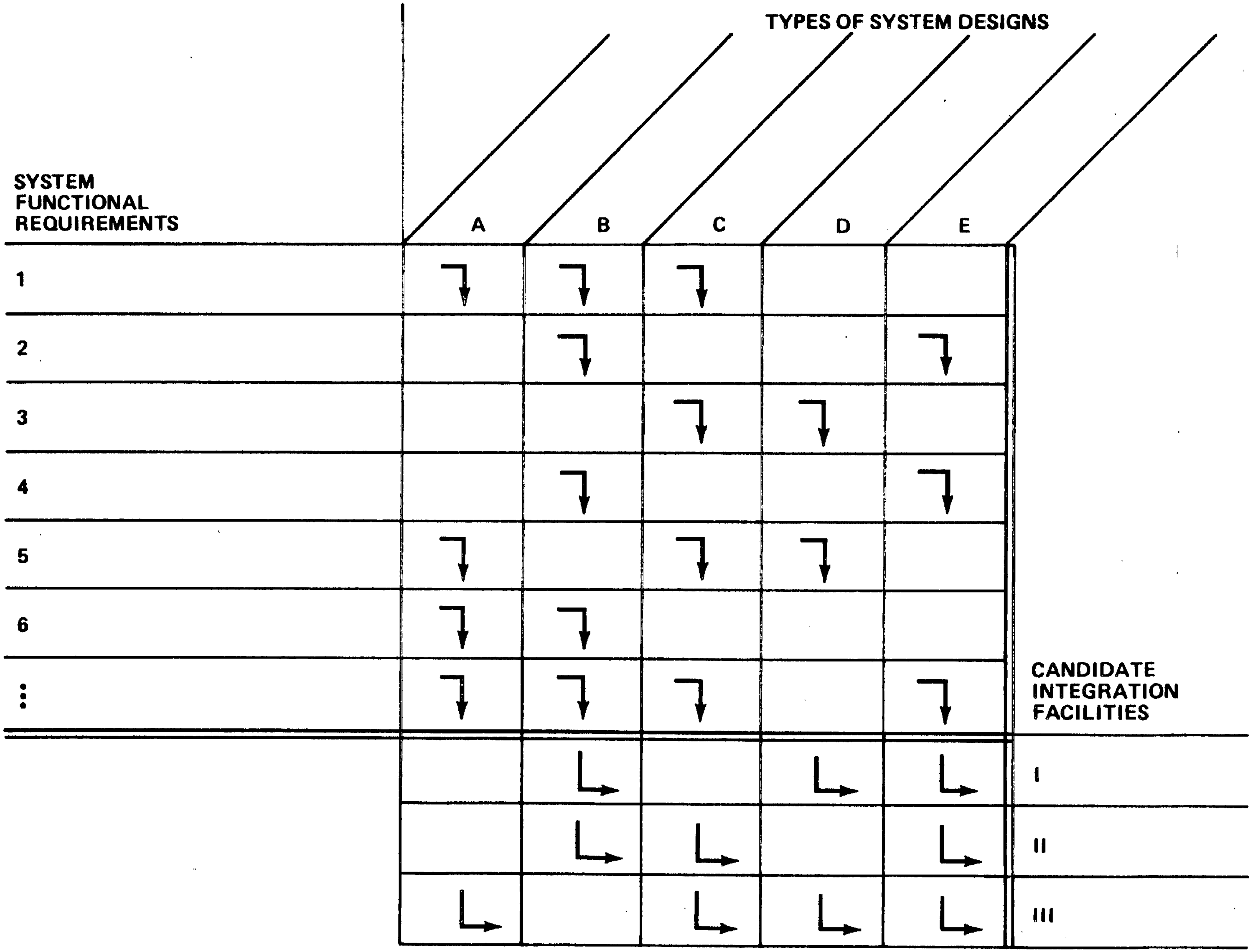


SYSTEM INTEGRATION GENERALITIES

- FACTORS INFLUENCING CRITICALNESS OF SYSTEM INTEGRATION
  - RISK TO CREW OR VEHICLE
  - RISK TO MISSION SUCCESS
  - UNTRIED TECHNOLOGY
  - CRITICAL TIMELINES FOR SYSTEM OPERATION
- FACTORS DETERMINING SYSTEM INTEGRATION FACILITY SELECTION
  - CAPABILITIES
  - AVAILABILITY
  - COST OF NEW FACILITY OR USAGE COST
- LARGE PROGRAMS
  - USUALLY HAVE LONG DEVELOPMENT PERIOD WITH SUSTAINING FUNDS
  - USUALLY DEVELOP THEIR OWN SYSTEM INTEGRATION FACILITIES
  - EXAMPLE – SHUTTLE AVIONICS INTEGRATION LAB
- MID SIZE PROGRAMS
  - USUALLY MODERATELY FUNDED AND MUST USE EXISTING INTEGRATION FACILITIES – WITH MODIFICATIONS AND ENHANCEMENTS
  - EXAMPLE – SPACE SHUTTLE MAIN ENGINE/HARDWARE SIMULATION LAB
- SHORT TERM PROJECTS
  - USUALLY MUST BE ACCOMMODATED BY EXISTING INTEGRATION FACILITIES
  - EXAMPLE – IMAGE MOTION COMPENSATION SYSTEM

SELECTION OF INTEGRATION FACILITIES

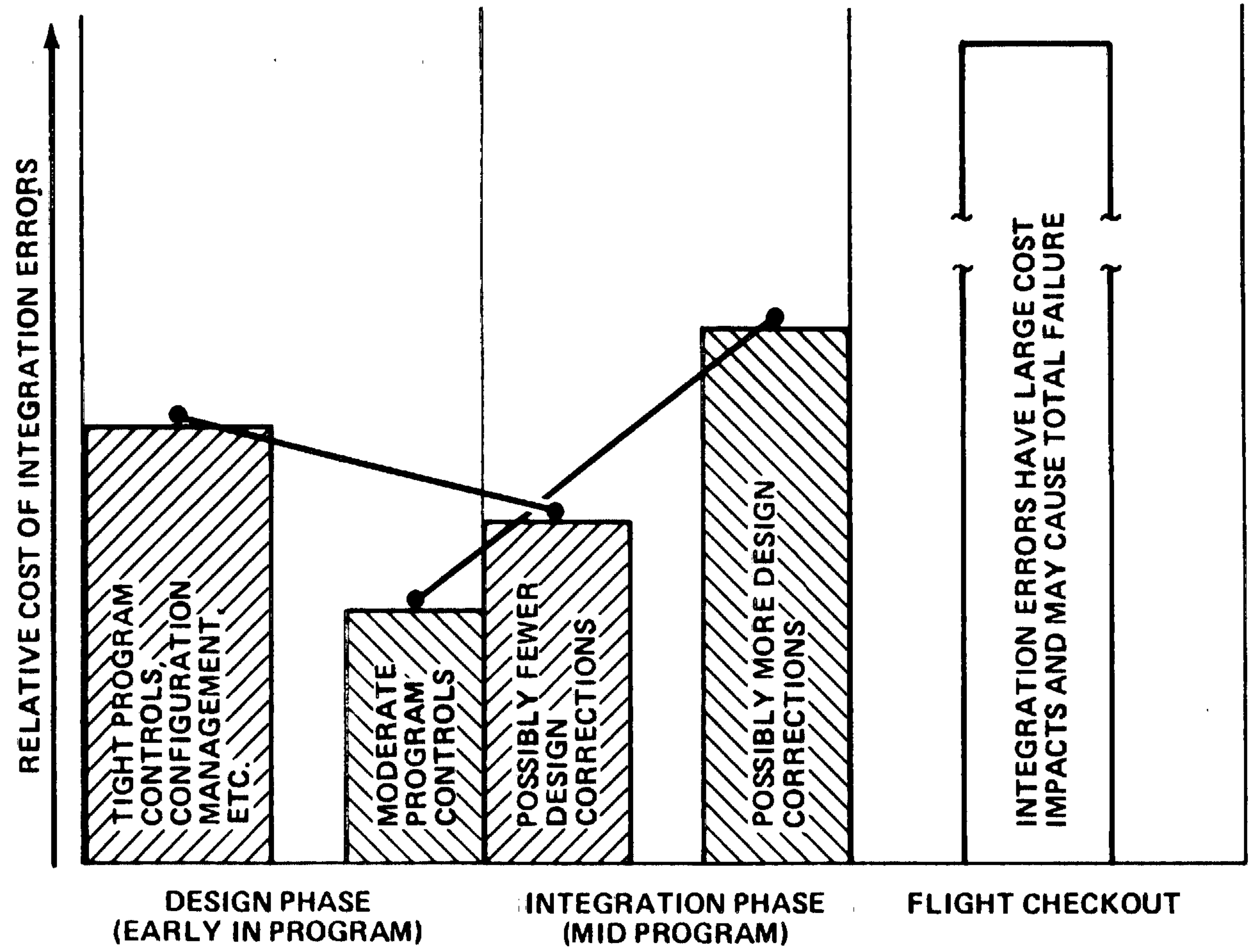


IDENTIFICATION OF SPACE STATION MECHANISM TEST BEDS

TYPES OF MECHANISMS

SPACE STATION FUNCTIONAL REQUIREMENTS	BERTHING	PROGRAMMABLE ACTUATORS	LATCHING DEVICES	MANIPULATORS	MECHANICAL JOINTS	
1. ASSEMBLY & EXCHANGE OF MAJOR ELEMENTS	↓	↓	↓			
2. ERECTION/DEPLOYMENT OF PANELS, RADIATORS, ETC.		↓	↓		↓	
3. ASSEMBLY OF STRUCTURAL BEAMS			↓	↓	↓	
4. ACTIVE SHAPE CONTROL OF LARGE DISTRIBUTED MEMBERS		↓				
5. SERVICING/REPAIR/ASSEMBLY BY FREE FLYER	↓		↓	↓		
6. ORBITER RESUPPLY MISSIONS	↓	↓				
7. CREW TRANSFER	↓	↓			↓	CANDIDATE SPACE STATION TEST BEDS
		→	→		→	SMART MECHANISM FACILITY
			→	→	→	AIR BEARING FLOOR
	→	→	→	→	→	COMPUTER DRIVER 6 DOF SYSTEM

COMPARISON OF SYSTEM INTEGRATION METHODS



SYSTEM INTEGRATION BY FACILITY EVOLUTION

DESIGN	COMPUTER ANALYSIS, TRADES, ETC. SYSTEM CONCEPT DEFINITION AND EVALUATION LARGE SCALE SIMULATION FOR DETAILED EVALUATION
SYSTEM INTEGRATION	ADDED FIDELITY BY INCORPORATION OF AVAILABLE BREADBOARD ITEMS FLIGHT SOFTWARE DEVELOPMENT INTEGRATION OF FLIGHT HARDWARE AND FLIGHT SOFTWARE
VERIFICATION & VALIDATION	OFF NOMINAL CONDITIONS FAILURE MODES REDUNDANCY MANAGEMENT CERTIFICATION OF FLIGHT SOFTWARE AND CRITICAL GROUND TEST SOFTWARE
MISSION SUPPORT	TRAINING OF GROUND CREWS AND FLIGHT CREW PRELAUNCH TESTING REAL TIME FLIGHT SUPPORT

EXAMPLE OF EVOLUTIONARY INTEGRATION FACILITY: SSME/HSL

TITLE - SYSTEM INTEGRATION METHODOLOGIES

PRESENTER - FRANK VINZ / MSFC

- SYSTEM INTEGRATION IS AN INTENSIVE AND TIME CRITICAL ACTIVITY WHICH MAY BE VULNERABLE TO DELAY AND UNIQUE SYSTEM PROBLEMS
- COMPLEXITY OF SYSTEM INTEGRATION INCREASES WITH THE AMOUNT OF SUBSYSTEM INTERFACES, THE NUMBER OF SEPARATE ORGANIZATIONS INVOLVED, AND THE NUMBER OF OPERATING OR REDUNDANT MODES OF THE SYSTEM
- PROGRAMS SHOULD BE MANAGED TO IDENTIFY AND CORRECT ERRORS AS EARLY AS POSSIBLE
- NEW TECHNOLOGIES SUCH AS FAULT TOLERANT SYSTEMS, SELF RECONFIGURABLE SYSTEMS, ARTIFICIAL INTELLIGENCE SYSTEMS, ETC SHOULD STRIVE TO REDUCE THE COMPLEXITIES OF SYSTEM INTEGRATION
- SYSTEM INTEGRATION IS AN VITAL OPERATION WHICH CANNOT BE TOTALLY BE ELIMINATED BY DESIGN CONTROLS
- COST REDUCTION OF SYSTEM INTEGRATION MAY BE POSSIBLE BY SHARING FACILITIES WITH DESIGN ACTIVITY OR EVOLVING FACILITIES FROM DESIGN TO SYSTEM INTEGRATION UTILIZATION

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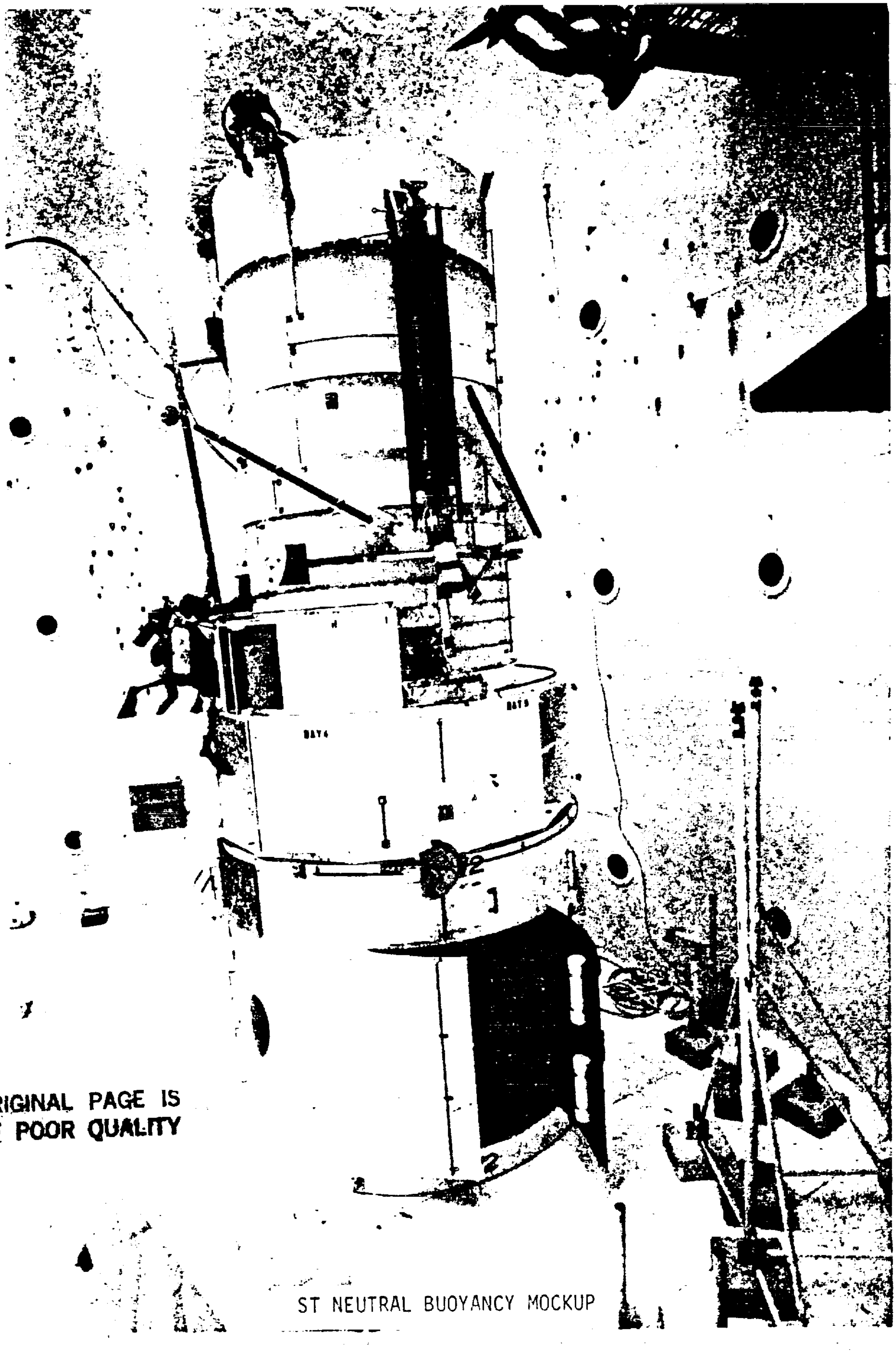
CREW SYSTEMS-ENGINEERING
SPACE TELESCOPE MAINTENANCE
AND REFURBISHMENT



ST NEUTRAL BUOYANCY MOCKUP

EL12

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CHART NO.:	CREW SYSTEMS, ST M&R	DATE: OCTOBER 1984

ABBREVIATIONS

EVA	EXTRA-VEHICULAR ACTIVITY
FGE	FINE GUIDANCE ELECTRONICS
FGS	FINE GUIDANCE SENSOR
FHST	FIXED HEAD STAR TRACKER
FSS	FLIGHT SUPPORT SYSTEM
LMSC	LOCKHEED MISSILES AND SPACE COMPANY
M&R	MAINTENANCE AND REFURBISHMENT
MFR	MANIPULATOR FOOT RESTRAINT
ORU	ORBITAL REPLACEMENT UNIT
PDR	PRELIMINARY DESIGN REVIEW
RGE	RATE GYRO ELECTRONICS
RMS	REMOTE MANIPULATOR SYSTEM
RSU	RATE SENSOR UNIT
RWA	REACTION WHEEL ASSEMBLY
S/A	SOLAR ARRAY
SI	SCIENTIFIC INSTRUMENT
SI C&DH	SCIENTIFIC INSTRUMENT COMMAND & DATA HANDLING
SSE	SPACE SUPPORT EQUIPMENT
ST	SPACE TELESCOPE
WF/PC	WIDE FIELD PLANETARY CAMERA

EL1:

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CHART NO.:	CREW SYSTEMS, ST M&R	DATE: OCTOBER, 1984

INTRODUCTION

SINCE LATE 1979 MSFC HAS BEEN DEVELOPING AN ON-ORBIT ST MAINTENANCE CAPABILITY. THE NUCLEUS OF THIS ACTIVITY HAS BEEN THE DEVELOPMENT OF DESIGNATED REPLACEABLE INSTRUMENTS, ST APPENDAGE OVERRIDE CAPABILITY, AND THE ASSOCIATED CREW AIDS FOR THESE TASKS. RECENTLY ST M&R EMPHASIS HAS SHIFTED TO DEVELOPMENT OF SPACE SUPPORT EQUIPMENT (SSE) TO BE CARRIED ABOARD THE STS IN SUPPORT OF A PLANNED MAINTENANCE MISSION. THIS PRESENTATION ADDRESSES THE CREW SYSTEMS DESIGN AND VERIFICATION EFFORT SUPPORTING PLANNED ST MAINTENANCE.

EL

ORGANIZATION: EL15	MARSHALL SPACE FLIGHT CENTER	NAME: JOHN REAVES
CHART NO.:	CREW SYSTEMS, ST M&R	DATE: OCTOBER 1984

ST CREW SYSTEMS APPROACH:

- BASED ON LMSC DESIGN/SCHEDULE CYCLES, EVALUATE ALL APPROPRIATE ST MECHANISMS FOR BOTH PLANNED AND CONTINGENCY MAINTENANCE
- THEN ATTENTION WAS TURNED TO FINAL DEFINITION OF CREW RELATED ST "SCARS". I.E. FOOT RESTRAINT SOCKETS AND HANDRAIL/HANDHOLDS
- ⇨ NEXT PRIORITY: SSE DESIGN SUPPORT EVALUATION

NOTE: CREW TRAINING HARDWARE WILL BE EVOLVED FROM DESIGN SUPPORT HARDWARE IN A TIMELY AND COST EFFECTIVE MANNER.

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CHART NO.:	**CREW SYSTEMS, ST M&R**	DATE: OCTOBER 1984

A TOTAL OF 11 NEUTRAL BUOYANCY SIMULATION SERIES HAVE BEEN CONDUCTED:

ST

AUG. '79	● EVALUATION OF SSM DOOR LATCHES, REMOVAL/REPLACEMENT OF FHST'S, RSU'S, & WF/PC, AND TOOL DEFINITION
NOV. '79	● RE–EVALUATION OF FHST/RSU ACCESS AND REMOVAL/REPLACEMENT OF AXIAL SI'S
DEC. '79	● EVALUATION OF SOLAR ARRAY AS AN ORU AND SOLAR ARRAY MECHANISMS ACCESS/OPERATION
FEB/MAR. '81	● DEFINITION AND EVALUATION OF FOOT RESTRAINT SOCKET LOCATIONS, HANDRAIL LOCATIONS, AND TOOLS FOR AXIAL/RADIAL SI CHANGEOUT AND SOLAR ARRAY CONTINGENCIES
MAY '81	● DEFINITION AND EVALUATION OF FOOT RESTRAINT SOCKETS, HANDRAILS, TOOLS, AND FOOT RESTRAINTS FOR CHANGEOUT OF FGS, RSU, FGE, RGE, SI C&DH, AND BATTERIES
JULY '81	● OVERALL EVA VERIFICATION SIMULATION BASED ON AGREEMENTS BETWEEN MSFC, JSC AND LMSC MADE AT JUNE, 1981 CREW TECHNICAL I/F MEETING
FEB. '84	● VERIFICATION OF ELECTRICAL UMBILICAL OVERRIDES, ACCESS TO CRYO VENTS AND BAYS 1, 4, 6, & 9

SSE (SPACE SUPPORT EQUIPMENT)

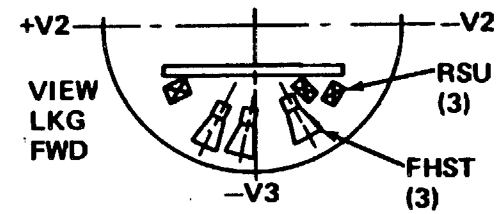
FEB. '82	● EVALUATION OF INITIAL SSE ORU CARRIER CONCEPTS STRESSING ACCESS TO MECHANISMS AND CREW AIDS LOCATIONS
APR/MAY '82	● EVALUATION OF REVISED ORU CARRIER UTILIZING RMS FOOT RESTRAINT FOR ACCESS AND TRANSFERS
FEB' '83	● EVALUATION OF RADIAL SI/FGS RAIL GUIDE EXTENSION CONCEPT
JULY '84	● ORU TRANSFER OPERATIONS (PDR CONCEPTS)

EL5061

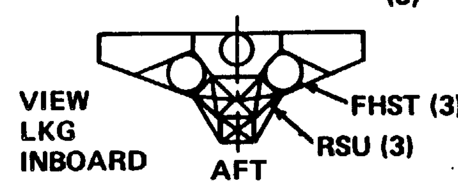
ORGANIZATION:		NAME:
EL15	MARSHALL SPACE FLIGHT CENTER **CREW SYSTEMS, ST M&R**	JOHN REAVES
		DATE: OCTOBER 1984

RATE SENSOR UNIT (RSU) REMOVAL/REPLACEMENT

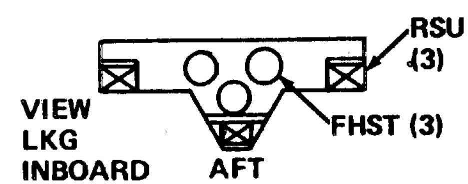
- INITIALLY THE PLACEMENT OF RSU'S MADE THEM INACCESSIBLE; THEY COULD NOT BE REPLACED ON ORBIT, AS DEMONSTRATED BY UNDERWATER SIMULATIONS



- LMSC PROPOSED SEVERAL ALTERNATIVES. THE MOST OBVIOUSLY WORKABLE SOLUTION WAS ALSO THE MOST EXPENSIVE TO IMPLEMENT



- BECAUSE OF THE HIGH COST OF THE PREFERRED APPROACH WE AGREED TO TRY ANOTHER LESS ATTRACTIVE SUGGESTION WHICH COULD BE IMPLEMENTED WITH NO COST IMPACT
- WE FOUND THE LESSER APPROACH WORKABLE (THOUGH NOT OPTIMUM) AND AGREED TO ACCEPT



WITHOUT THE OPPORTUNITY TO TEST, WE WOULD HAVE HAD TO RECOMMEND THE APPROACH WHICH ELIMINATED ALL QUESTION – THE MOST EXPENSIVE

EL12

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CHART NO.:	CREW SYSTEMS, ST M&R	DATE: OCTOBER 1984

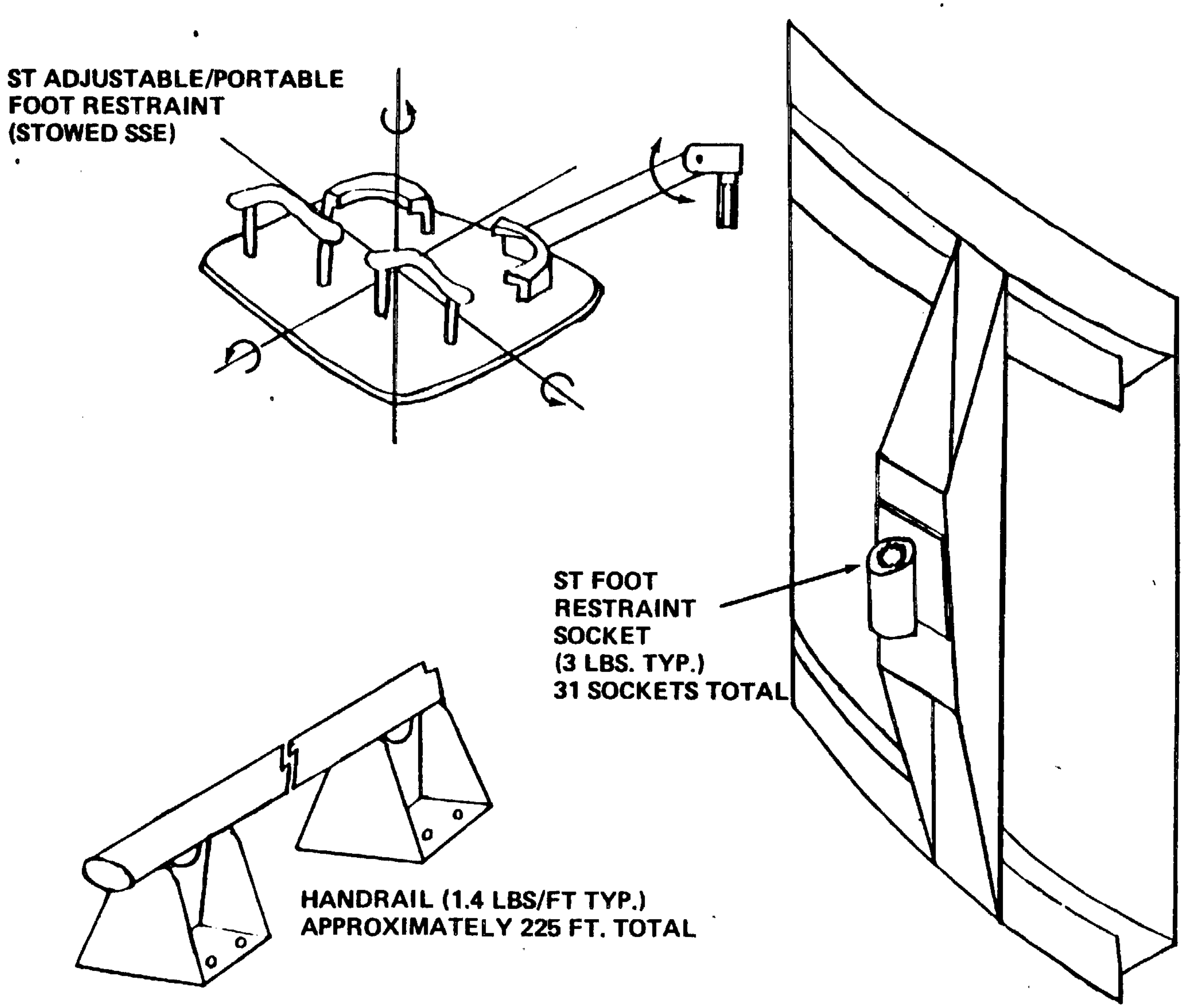

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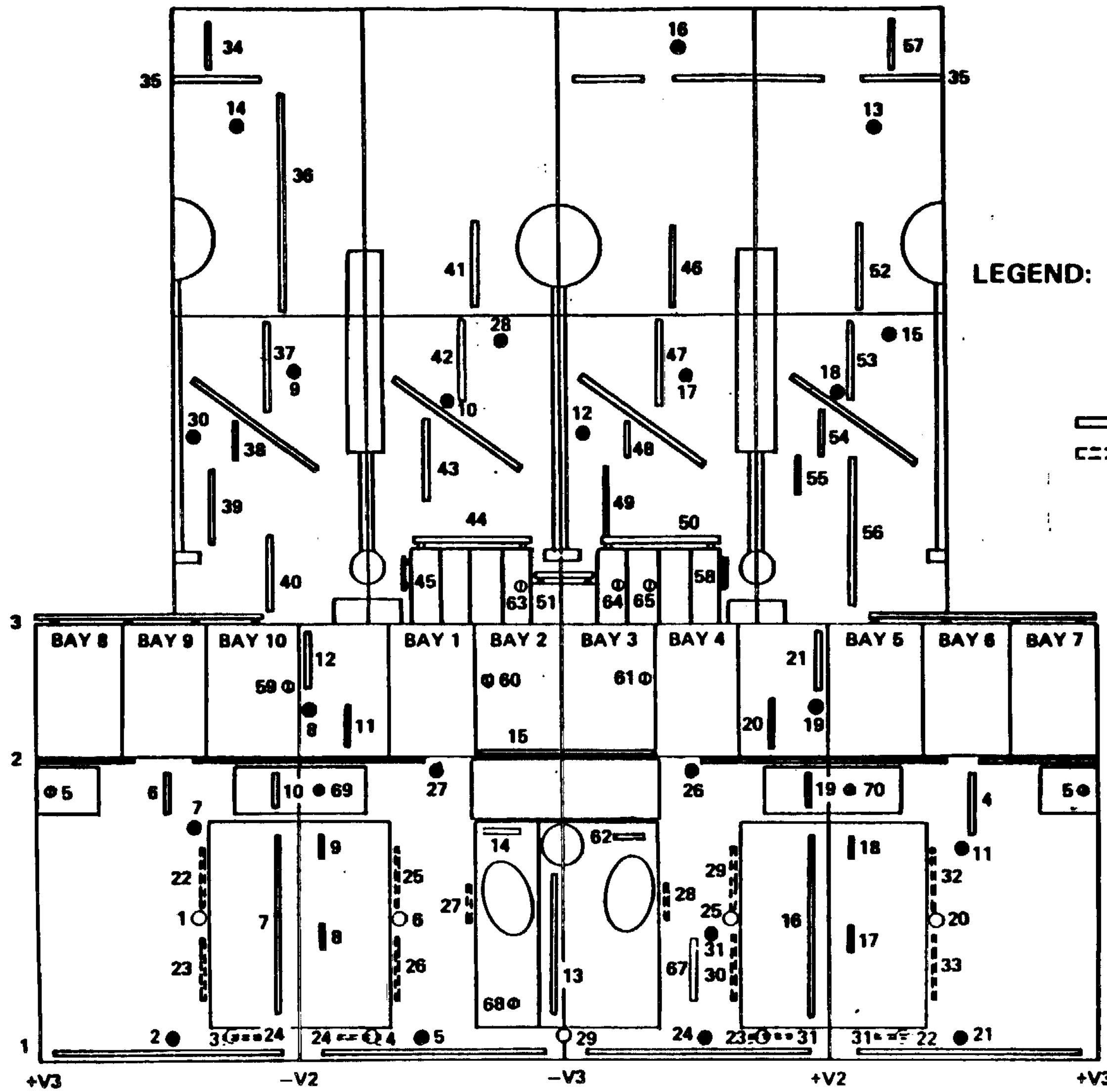
CHART NO.:	**CREW SYSTEMS, ST M&R**	DATE: OCTOBER 1984



LEGEND: FOOT RESTRAINT SOCKETS
- ● EXTERIOR
- O INTERIOR
- HAND KNOB ⊕
- HANDRAIL
  - EXTERIOR
  - INTERIOR

EL1 OI ZATION: EL15	MARSHALL SPACL ;HT CENTER	NAME: JOHN REAVES
CHART NO.:	CREW SYSTEMS, ST M&R	DATE: OCTOBER 1984

PRELIMINARY
FOOT RESTRAINT DESIGN

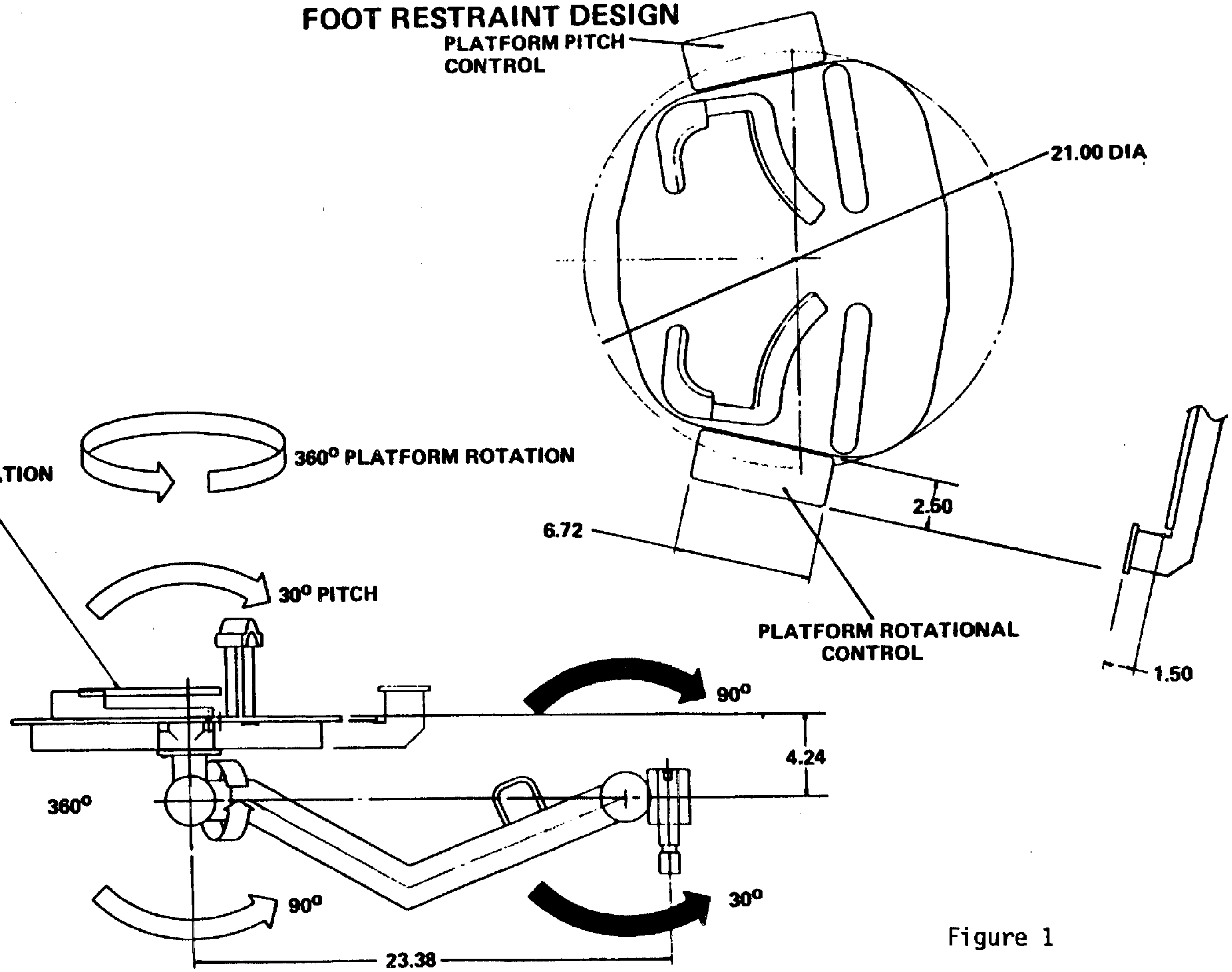


Figure 1

ST EVA RATCHET WRENCH EXTENSIONS

DIMENSION A (IN)

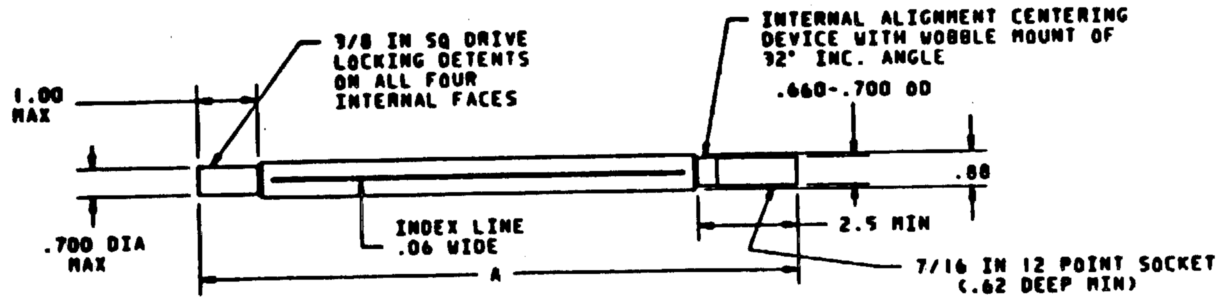
- • 1 - 24.0
- • 2 - 18.0
- • 3 - 12.0
- • 4 - 6.0
- • 5 - 2.0 SOCKET ONLY W/.18 DIA TETHERED PIN WITH RING

• INDICATES REQUIRED FOR DEPLOYMENT MISSION

EVA RATCHET WRENCH TORQUE LIMITER

ST TORQUE REQUIREMENTS:

40-42 FT-LBS AXIAL SI/RADIAL SI/FGS
7.5-9.0 FT-LBS ORU BOLTS
• 5.3-6.5 FT-LBS SOLAR ARRAY



NOTE:
FINISH-CLEAR ANODIZE WITH BLACK MARKINGS
7.0 IN-OZ MAX. RATCHETING TORQUE

EVA RATCHET WRENCH

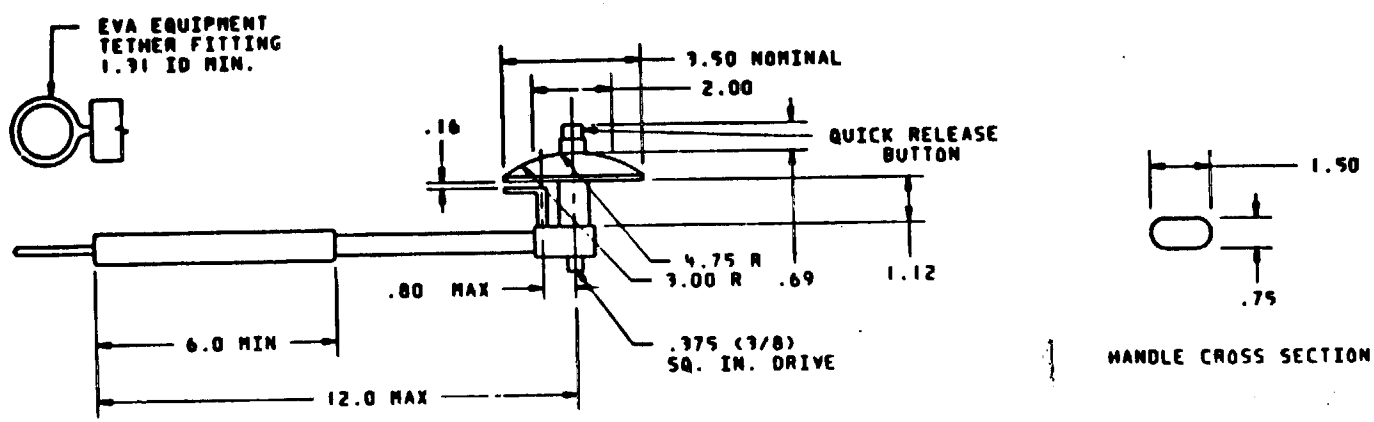


FIGURE 2 - EVA RATCHET WRENCH AND EXTENSION

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CHART NO.:	CREW SYSTEMS, ST M&R	DATE: OCTOBER 1984

ORU FASTENER DESIGNS:

1. J-HOOK - FIGURE 3
2. CAPTIVE STUD - FIGURE 4
3. KEYWAY - FIGURE 5

NOTE: ALL ORU BOLT INTERFACES ARE 7/16 INCH HEX OR 12 POINT.

ORU ELECTRICAL CONNECTOR DESIGNS:

1. WING TAB - FIGURE 4
2. DRIVE PLATE - FIGURE 5

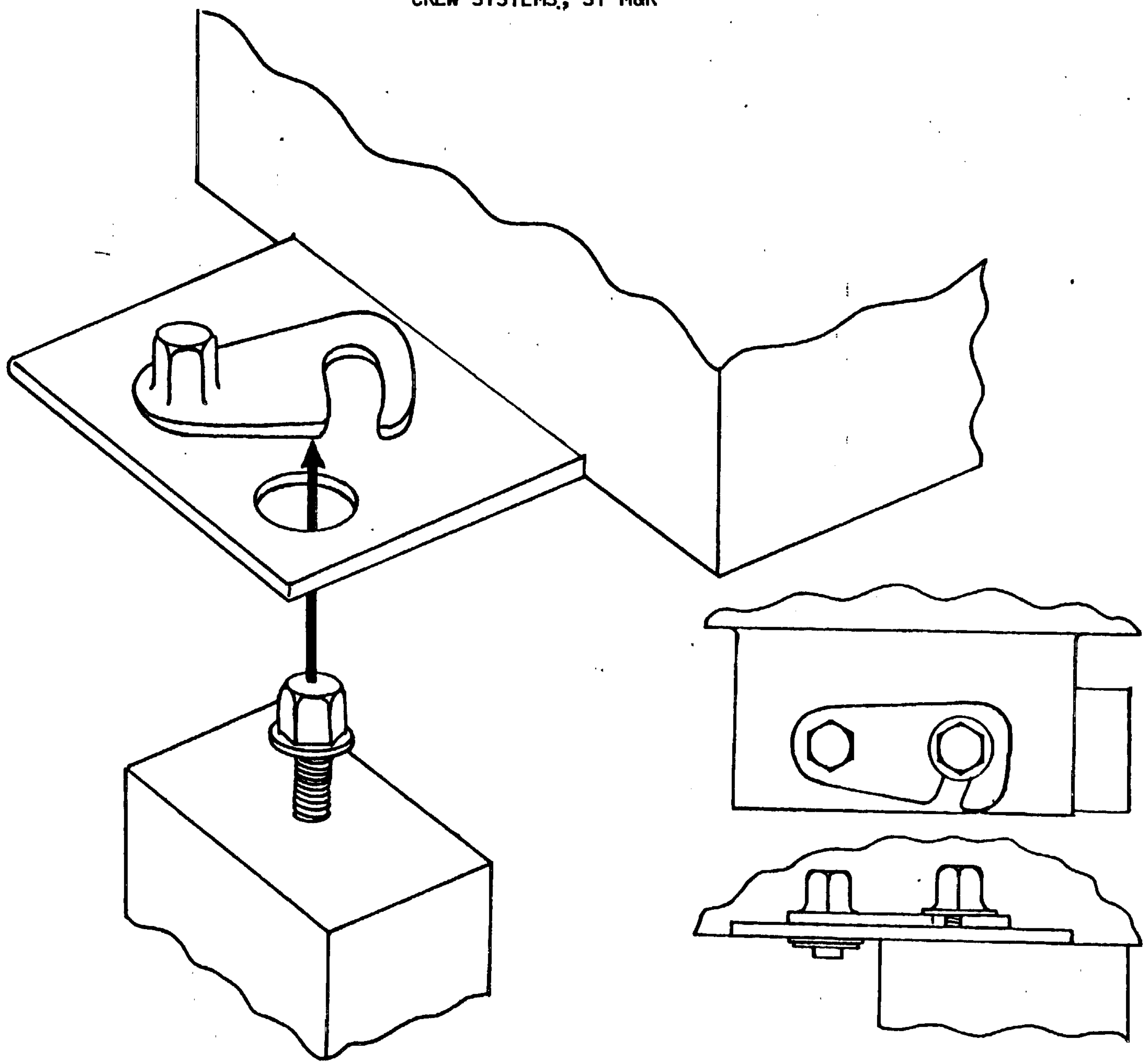
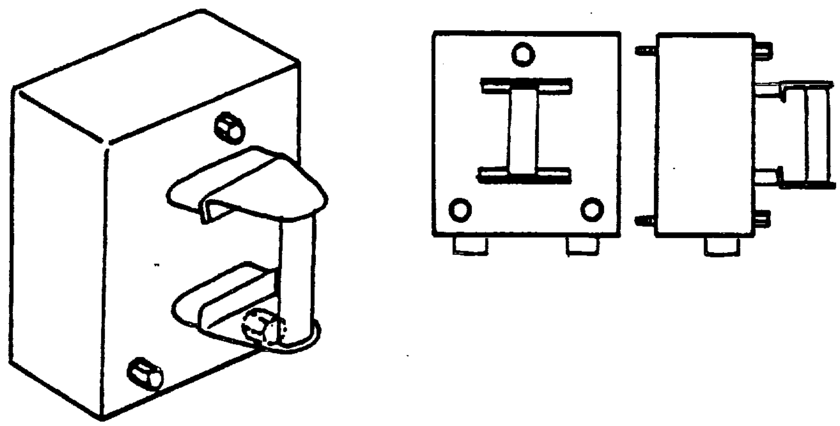


Figure 3 - J-hook latch (e.g., Batteries, DF224 Computer)



RSU mounting bolt locations.

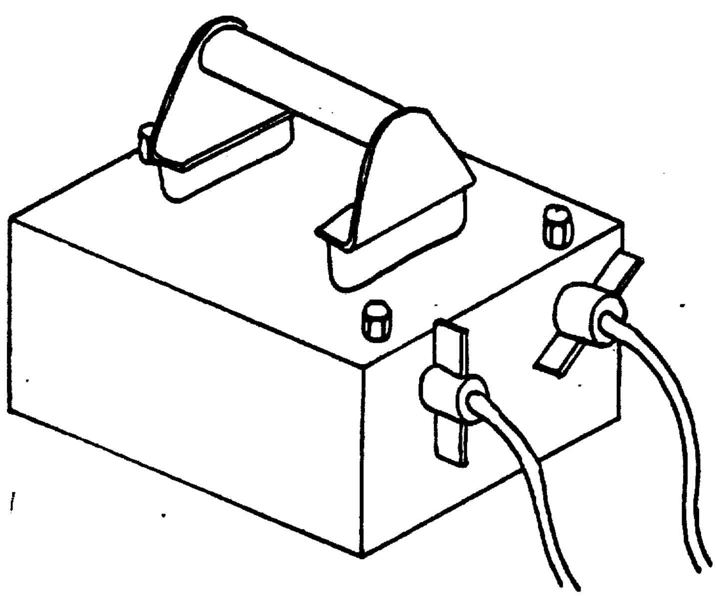


Figure 4 - RSU electrical wing tab connectors.

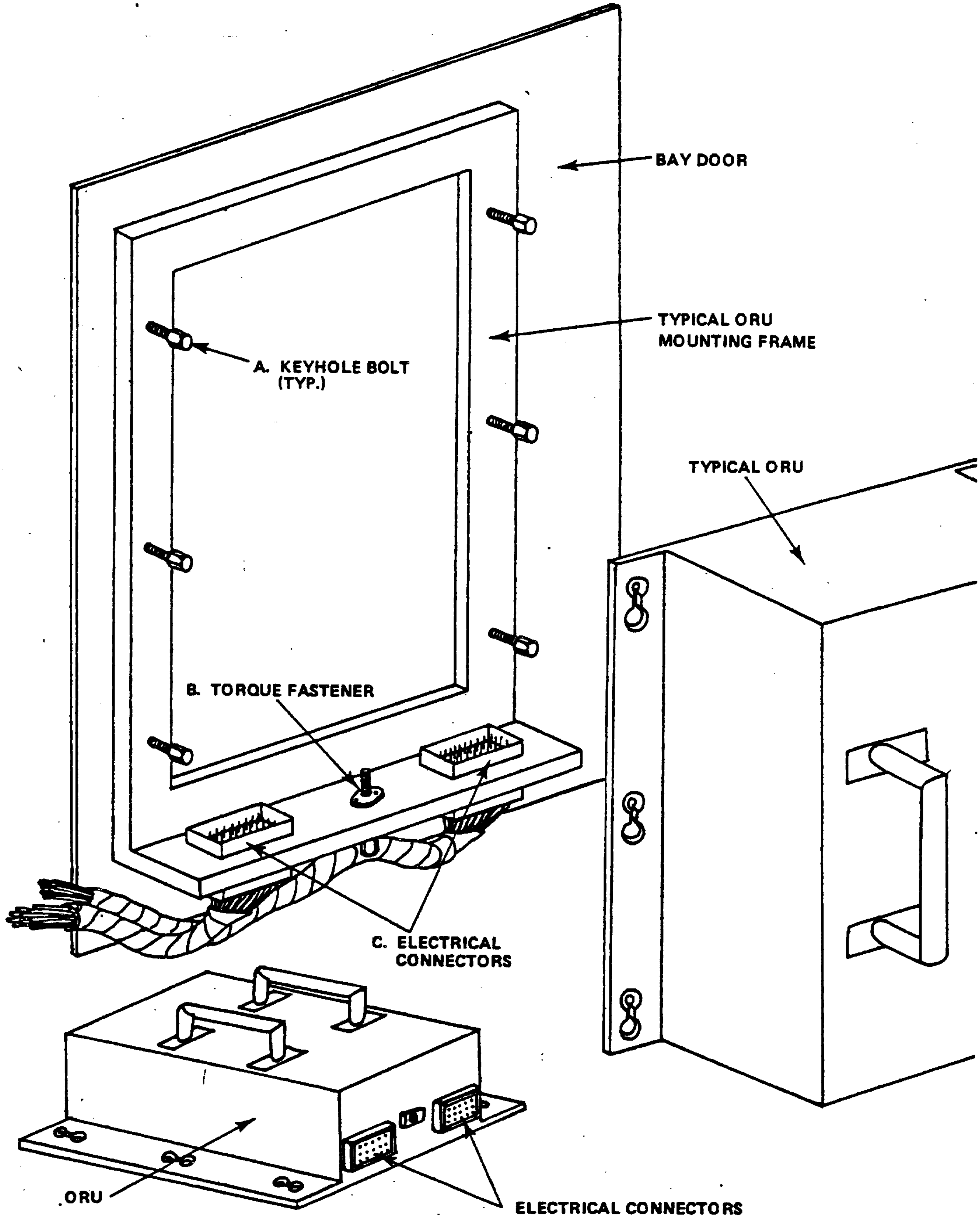


Figure 5 - Typical ORU (e.g., SI C&DH) door mounting system.

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- BATTERY INTERFACES VERIFIED
  - LOCATION - BAYS 2 & 3 ON DOOR (6 UNITS)
  - ACCESS - VIA FOOT RESTRAINT SOCKETS 26 & 27
  - MECHANISMS - (EACH BATTERY)
    - 6 ORU BOLTS, J-HOOK TYPE
    - 2 WING TAB ELECTRICAL CONNECTORS

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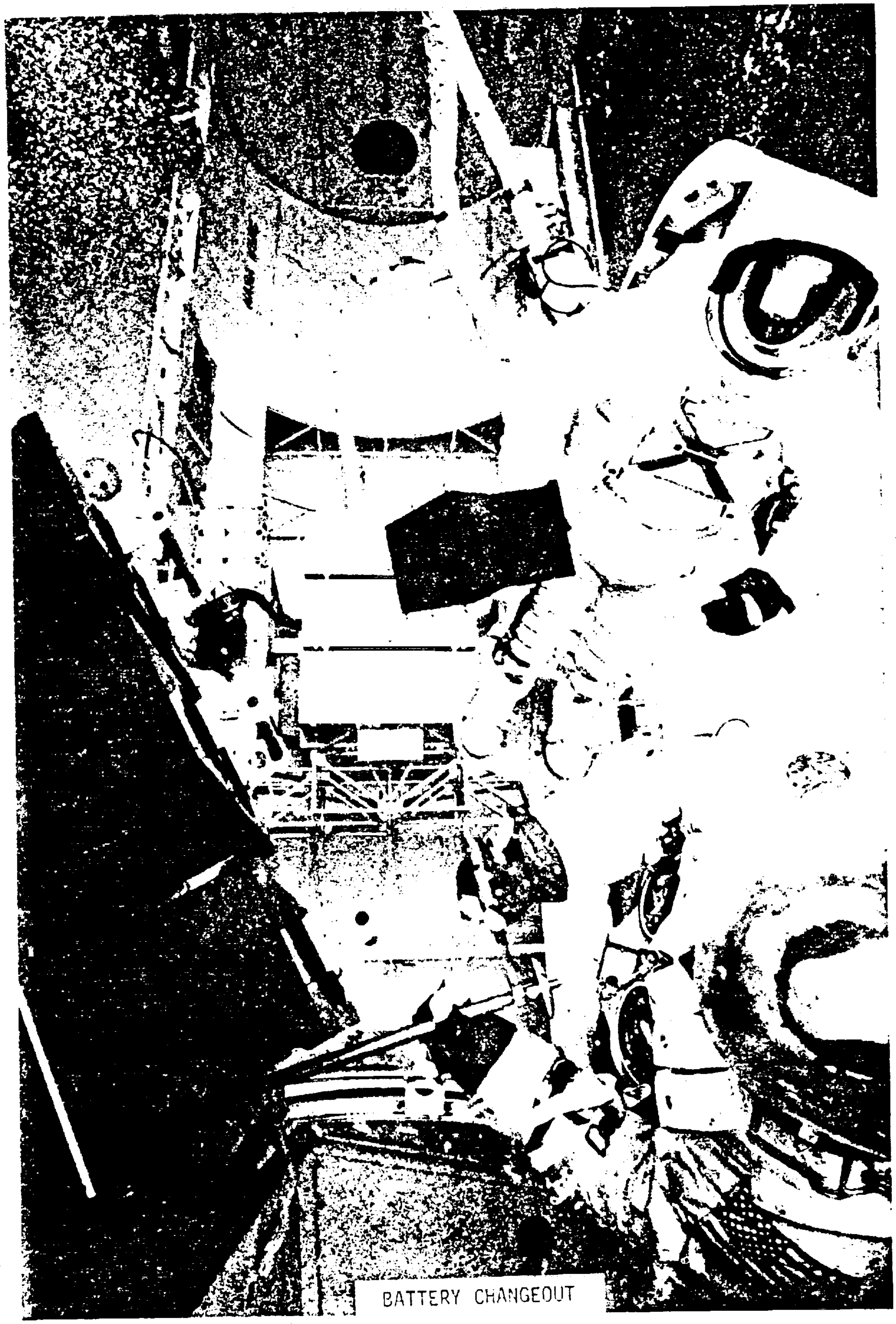

BATTERY CHANGEOUT

ORGANIZATION: EL15	MARSHALL SPACE FLIGHT CENTER	NAME: JOHN REAVES
CHART NO.:	CREW SYSTEMS, ST M&R	DATE: OCTOBER 1984

- RATE SENSING UNIT (RSU) INTERFACES VERIFIED
  - LOCATION – FIXED HEAD STAR TRACKER (FHST) BAY EQUIPMENT SHELF (3 UNITS)
  - ACCESS – VIA FOOT RESTRAINT SOCKET 29
  - MECHANISMS – (EACH RSU)
    - 3 ORU BOLTS, J-HOOK TYPE ON FHST SHADES
    - 3 ORU BOLTS, STANDARD CAPTIVE TYPE ON RSU
    - 2 WING TAB ELECTRICAL CONNECTORS

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RSU CHANGEOUT

ORGANIZATION: EL15	MARSHALL SPACE FLIGHT CENTER	NAME: JOHN REAVES
CHART NO.:	**CREW SYSTEMS, ST M&R**	DATE: OCTOBER 1984

- SCIENTIFIC INSTRUMENT COMMAND AND DATA HANDLING (SIC & DH) INTERFACES VERIFIED
  - LOCATION - BAY 10 ON DOOR (1 UNIT)
  - ACCESS - VIA FOOT RESTRAINT SOCKET 7
  - MECHANISMS -
    - 4 ORU BOLTS, KEYWAY TYPE
    - 6 ORU BOLTS, STANDARD CAPTIVE TYPE
    - 1 ELECTRICAL CONNECTOR DRIVE BOLT

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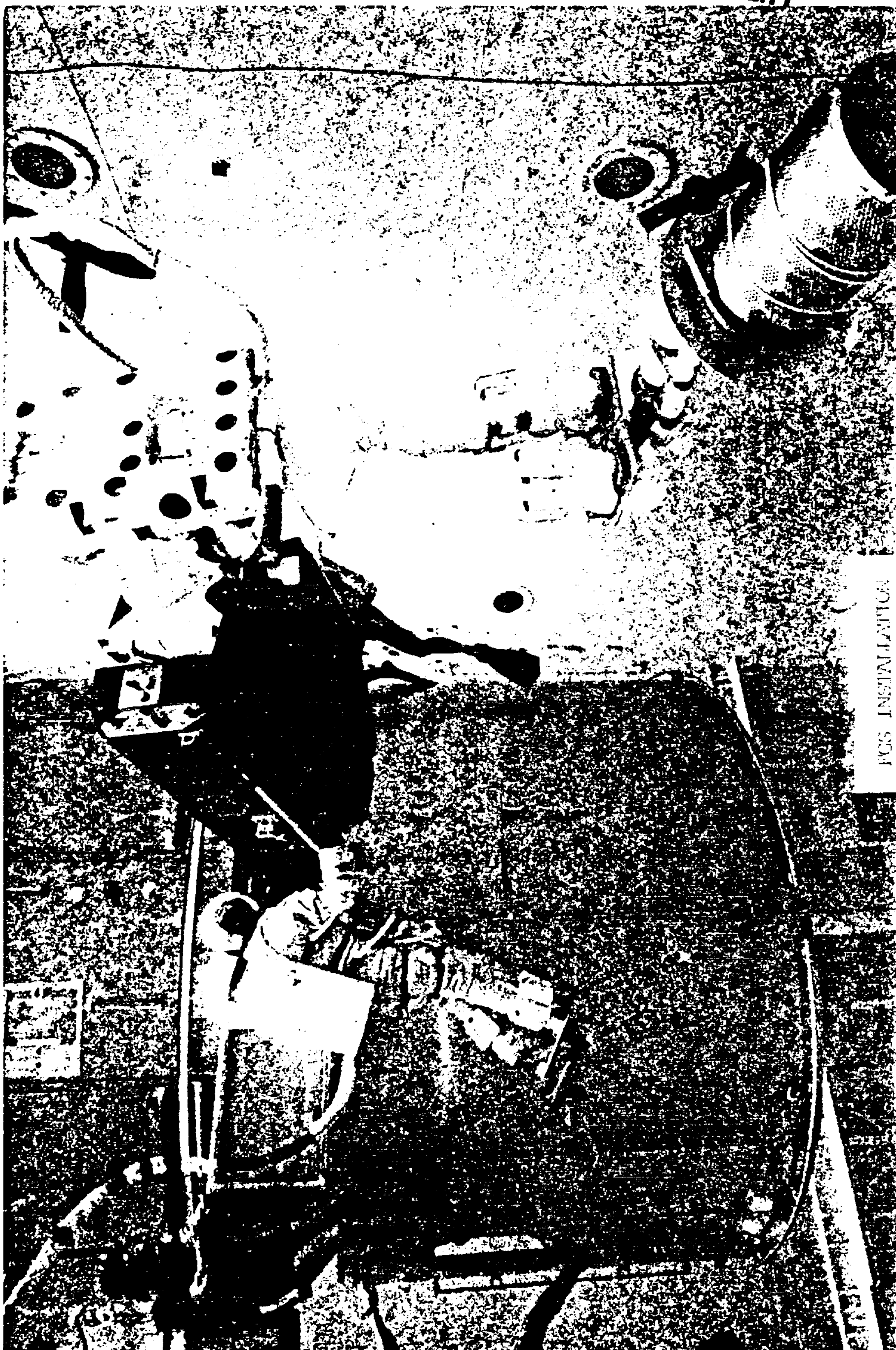
SI C&DH CHANGEOUT

ORGANIZATION: EL15	MARSHALL SPACE FLIGHT CENTER	NAME: JOHN REAVES
CHART NO.:	**CREW SYSTEMS, ST M&R**	DATE: OCTOBER 1984

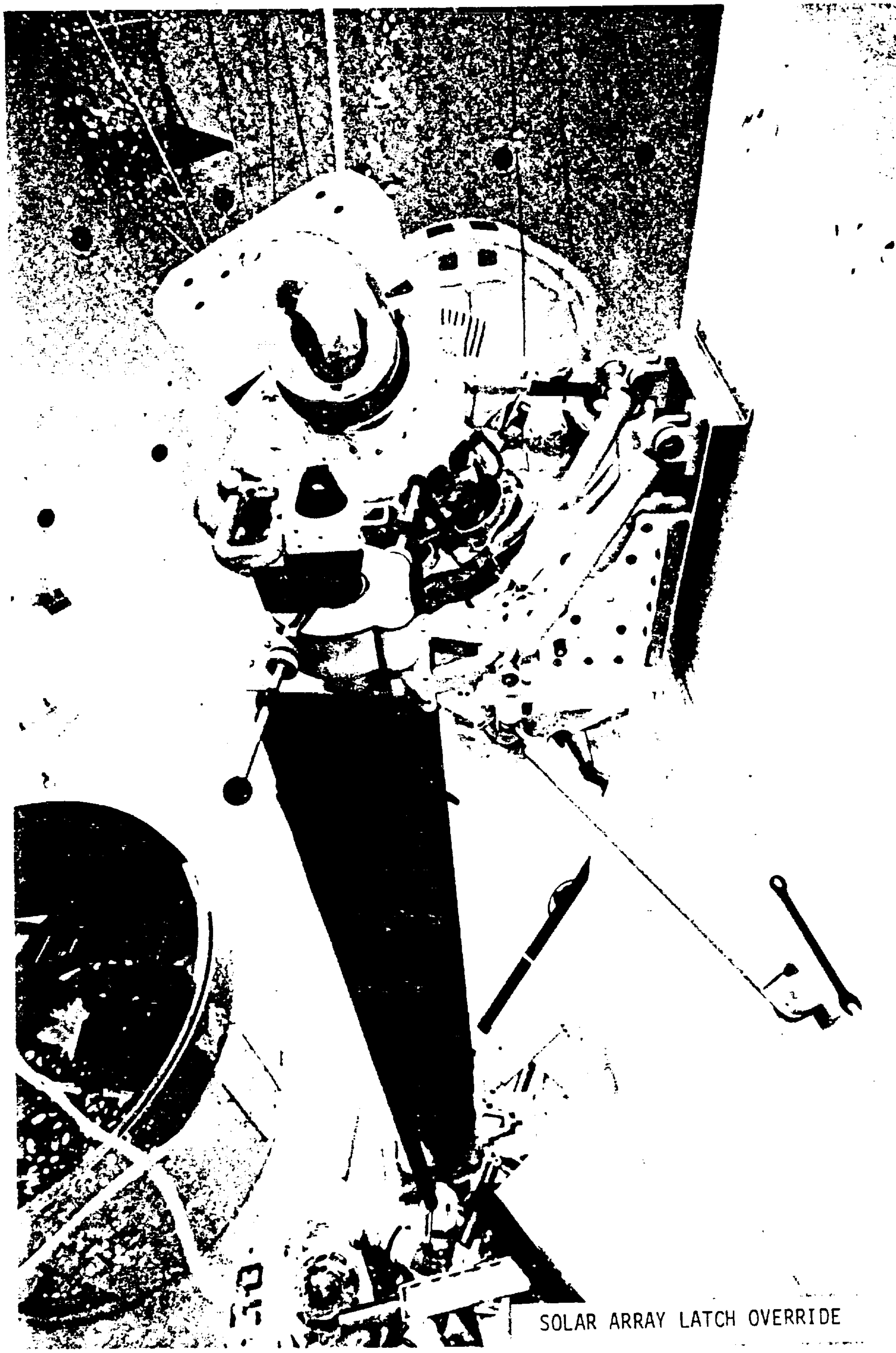
- FINE GUIDANCE SENSOR (FGS) INTERFACES VERIFIED
  - LOCATION - RADIAL BAYS ± V_2 AXIS + V_3 AXIS (3 UNITS)
  - ACCESS - VIA FOOT RESTRAINT SOCKETS 7, 11, 26, 27, & 31
  - MECHANISMS - (EACH FGS)
    - 'A' LATCH INTERFACE
    - ACCESS TO 9 WING TAB ELECTRICAL CONNECTORS
    - GUIDERAIL INTERFACE



PCS INSTALLATION

- SOLAR ARRAY (S/A)
  - LOCATION – ± V_2 AXIS (2 UNITS)
  - ACCESS – VIA FOOT RESTRAINT SOCKETS 8, 9, 10 AND 17, 18, 19
  - MECHANISMS – OVERRIDE THE FOLLOWING:
    (EACH ARRAY)
    - FORWARD/AFT S/A LATCHES
    - PRIMARY DRIVE MECHANISM (PDM)
    - SECONDARY DRIVE MECHANISM (SDM)
    - OFF LOAD DEVICE (OLD)
    - BRAKES
    - SOLAR ARRAY DRIVE (SAD)

    JETTISON PROVISIONS:
    - JETTISON CLAMP
    - DIODE BOX CONNECTORS
    - JETTISON HANDLE/GRAPPLE FIXTURE INTERFACE



SOLAR ARRAY LATCH OVERRIDE

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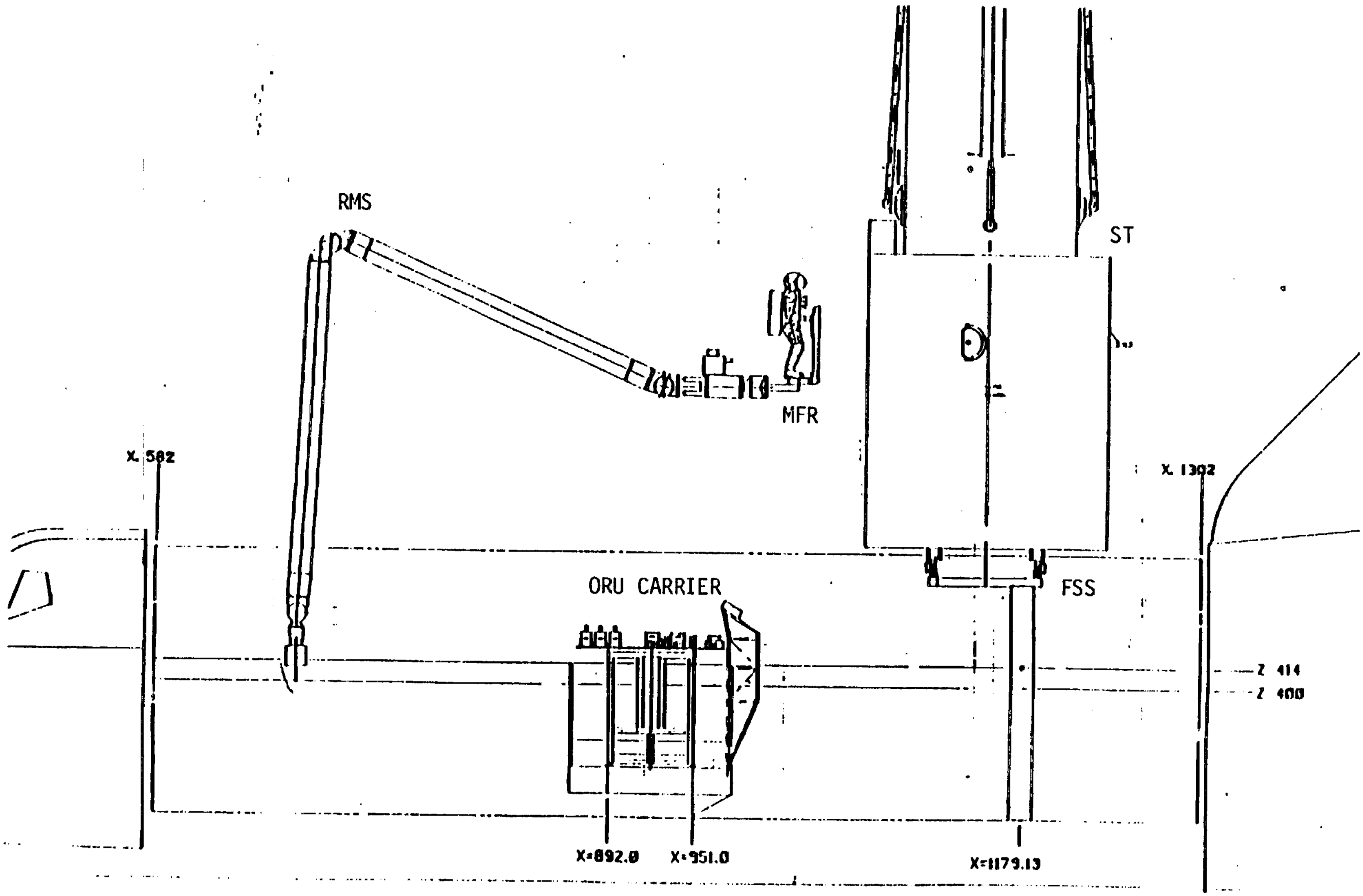
CHART NO.:	**CREW SYSTEMS, ST M&R**	DATE: OCTOBER 1984

- INITIAL EVALUATIONS OF ORBITAL REPLACEMENT UNIT (ORU) CARRIER CONCEPTS
    - WITH ST DOCKED TO FLIGHT SUPPORT SYSTEM (FSS), PERFORM INSTRUMENT CHANGEOUTS
    - EVALUATED WORK STATIONS AND TRANSFER ROUTES ON CARRIER
    - EVALUATED CARRIER MECHANISMS AND STOWAGE CONCEPTS
    - EVALUATED RMS WITH FOOT RESTRAINTS FOR ACCESS TO CARRIER, TRANSFER OF INSTRUMENTS AND INSTALLATIONS ON ST

CREW SYSTE ST M&R

MAINTENANCE CONFIGURATION



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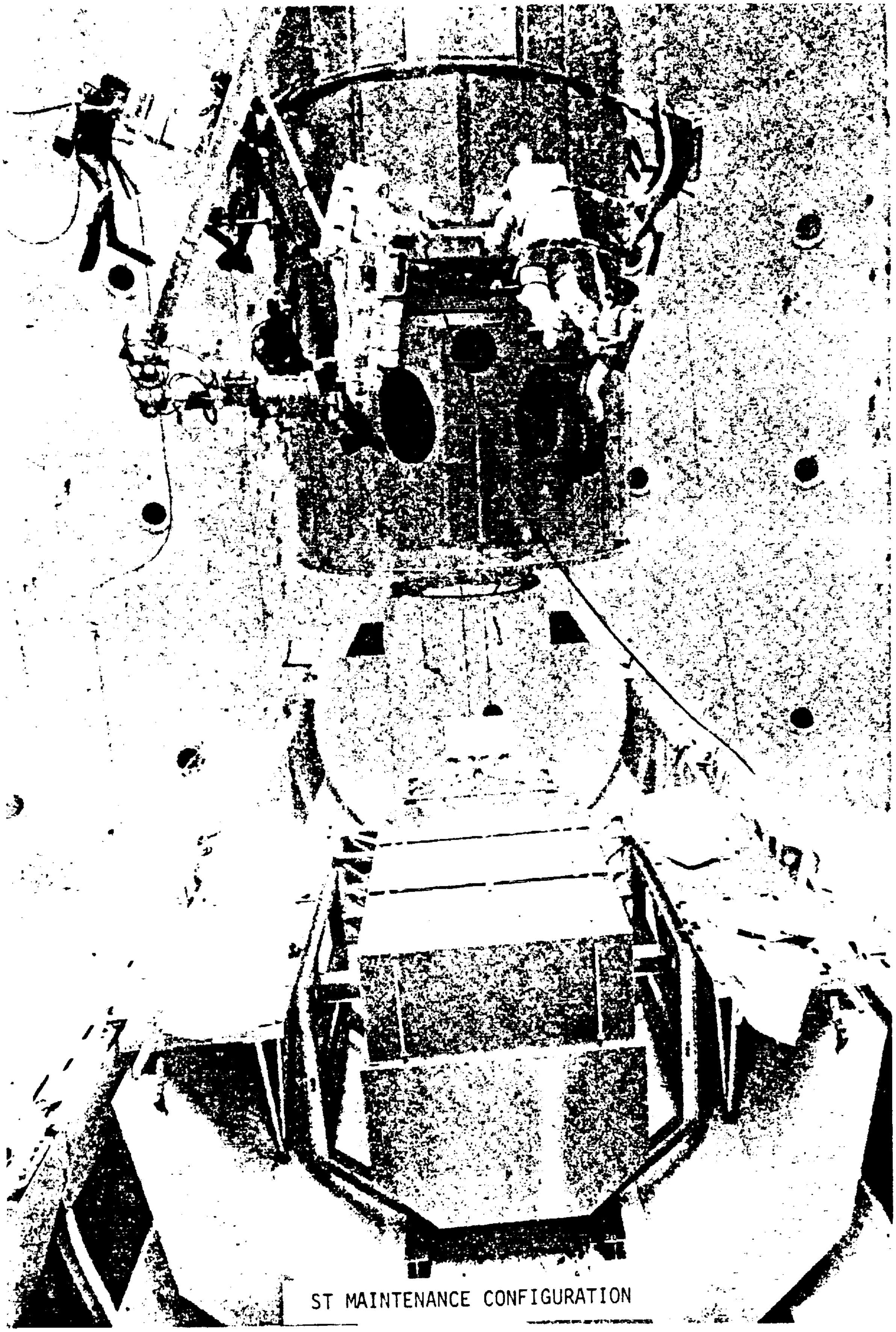

ST MAINTENANCE CONFIGURATION

ITEM	TOTAL ST COMPLEMENT	MAINTENANCE MISSION COMPLEMENTS (NOTE 1)			
		OPTION #A	OPTION #B	OPTION #C	OPTION #D
BATTERIES	6	6	6	6	6
SOLAR ARRAY	2	0	0	0	2
FINE GUIDANCE SENSOR	3	1	1	UP TO 3	0
FINE GUIDANCE ELECTRONICS	3	1	1	UP TO 3	0
DF244 COMPUTER	1				1
SI C&DH	1	ANY ONE	ANY ONE	ANY ONE	1
RWA	4				2
RATE SENSOR UNIT	3	1	1	1	1
RATE GYRO ELECTRONICS	3	1	1	1	1
FUSE PLUG	12	12	12	12	12
WIDE FIELD PLANETARY CAMERA	1	1	0	0	0
FAINT OBJECT CAMERA	1	0		0	0
HIGH SPEED PHOTOMETER	1	0		0	0
HIGH RESOLUTION SPECTROGRAPH	1	0	ANY ONE	0	0
FAINT OBJECT SPECTROGRAPH	1	0		0	0

NOTE 1: ORU COMPLEMENTS USED FOR SIZING THE ORU CARRIER.

TABLE 1 - ORU LIST

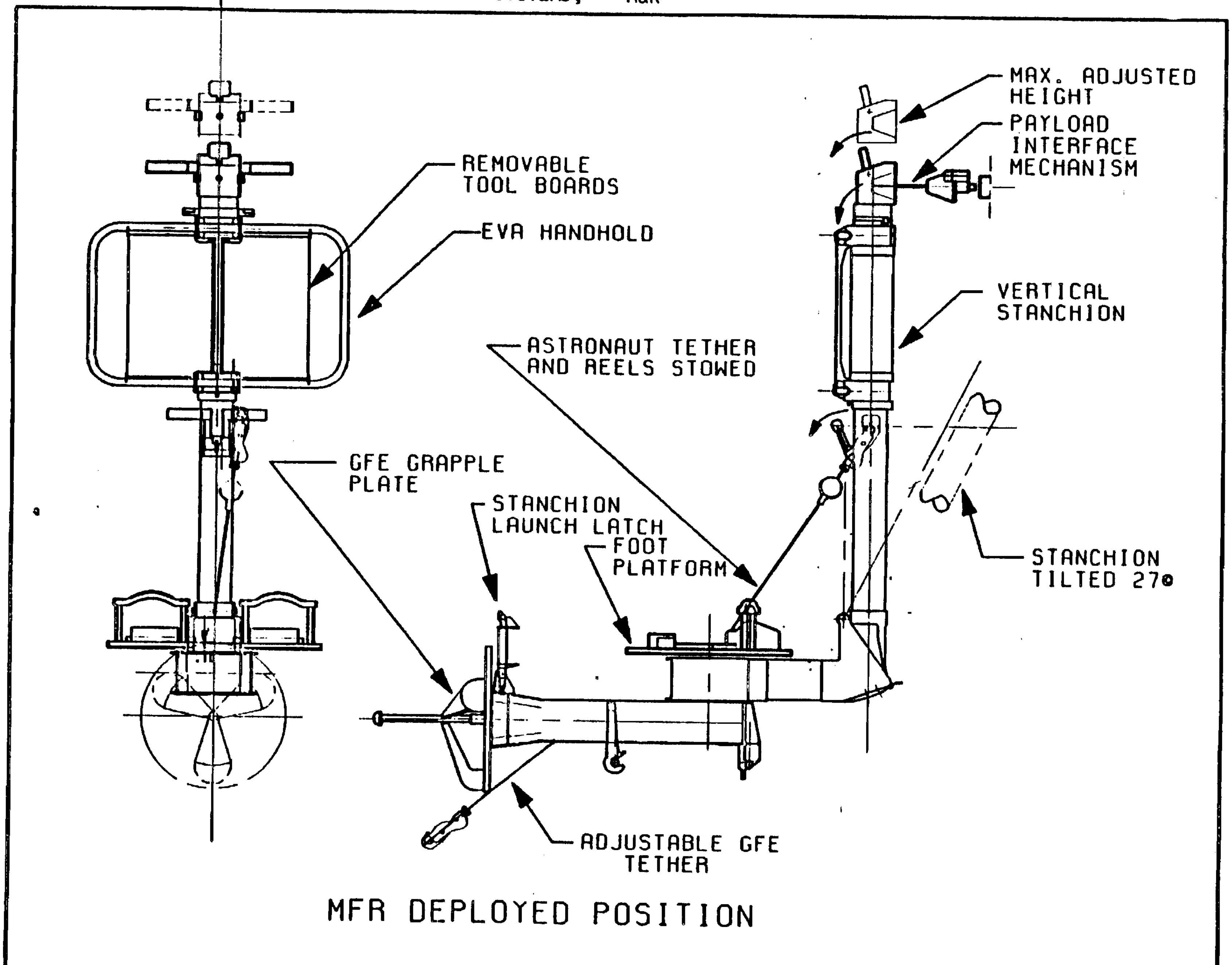
MAX. ADJUSTED HEIGHT
PAYLOAD INTERFACE MECHANISM
REMOVABLE TOOL BOARDS
EVA HANDHOLD
VERTICAL STANCHION
ASTRONAUT TETHER AND REELS STOWED
GFE GRAPPLE PLATE
STANCHION LAUNCH LATCH
FOOT PLATFORM
STANCHION TILTED 27°
ADJUSTABLE GFE TETHER
MFR DEPLOYED POSITION

Figure 7

MFR - BATTERY TRANSFER

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CHART NO.:	CREW SYSTEMS, ST M&R	DATE: OCTOBER 1984

SUMMARY

PLANNED MAINTENANCE DEVELOPMENT ON THE SPACE TELESCOPE HAS EVOLVED MANY INNOVATIVE EVA HARDWARE ITEMS AND OPERATIONAL TECHNIQUES. FUTURE WORK REMAINS IN THE AREA OF DESIGNING AN ORBITAL REPLACEMENT UNIT (ORU) CARRIER THAT IS COMPATIBLE WITH THE STS, THE ST/FSS, AND THE EVA CREW. THE FOLLOWING CONCLUSIONS/RECOMMENDATIONS PROVIDE A SOUND BASIS FOR THE CONTINUATION OF THIS DEVELOPMENT EFFORT.

EL12

ORGANIZATION: EL15	MARSHALL SPACE FLIGHT CENTER	NAME: JOHN REAVES
CHART NO.:	CREW SYSTEMS, ST M&R	DATE: OCTOBER 1984

CONCLUSIONS/RECOMMENDATIONS

1. STANDARDIZATION OF ORBITAL REPLACEMENT UNIT (ORU) FASTENER DESIGNS AND ELECTRICAL CONNECTOR DESIGNS GREATLY ENHANCES THE EVA CHANGE-OUT TASK.

2. A UNIQUE SET OF EVA TOOLS IS BEING DEVELOPED TO MEET SPECIFIC ST MAINTENANCE REQUIREMENTS.

3. A POWER RATCHET SHOULD BE PROVIDED AS A PRIMARY TOOL WITH THE MANUAL RATCHET AS BACKUP. A POWER RATCHET IS MANY TIMES FASTER AND GREATLY REDUCES HAND/ARM FATIGUE.

5. THE REMOTE MANIPULATOR SYSTEM (RMS) WITH THE MANIPULATOR FOOT RESTRAINT (MFR) ATTACHED PROVIDES AN EVEN MORE FLEXIBLE SYSTEM SINCE IT PROVIDES A WORK PLATFORM WITH TOOL BOARD AND PAYLOAD ATTACH PROVISIONS AND SERVES AS AN ORU TRANSFER DEVICE.

6. THE FLIGHT SUPPORT SYSTEM (FSS) (CONFIGURED TO SUPPORT ST PLANNED MAINTENANCE) MEETS ALL ST BERTHING AND TENDING REQUIREMENTS.

JOHN REAVES - RENDEZVOUS AND PROX OPS WORKSHOP

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4. THE ST PORTABLE FOOT RESTRAINT (ADJUSTABLE IN ELEVATION, PITCH, ROLL, AND YAW), IN CONJUNCTION WITH STRATEGICALLY LOCATED MOUNTING SOCKETS, PROVIDES A VERSATILE EVA WORK PLATFORM.
5. THE REMOTE MANIPULATOR SYSTEM (RMS) WITH THE MANIPULATOR FOOT RESTRAINT (MFR) ATTACHED PROVIDES AN EVEN MORE FLEXIBLE SYSTEM SINCE IT PROVIDES A WORK PLATFORM WITH TOOL BOARD AND PAYLOAD ATTACH PROVISIONS AND SERVES AS AN ORU TRANSFER DEVICE
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SYSTEM INTEGRATION/ SIMULATION TESTS

A. Nathan

GRUMMAN AEROSPACE CORPORATION

GRUMMAN

Phasing Integration/Simulation Capabilities

The size and complexity of the Space Station program warrants planned phasing of simulators, verification facilities and trainers to achieve economies in life cycle costs. The facing page illustrates some of these phasing options. One option is to phase a development simulator into a verification facility and ultimately then into a trainer.

This approach may not be feasible in the case of Space Station because of its multimission role and the need to grow capabilities with time. As illustrated, the Trainer would have to be used as a development facility during the operational phase of the initial system.

This paper reports development simulator capabilities and past experiences at Grumman as it pertains to proximity operations. Because of the broad spectrum of developments required in Space Station, the concept of retaining purely development simulators is supported.

PHASING INTEGRATION/SIMULATION CAPABILITIES

- DEVELOPMENT PHASE
  - TEST & EVALUATE OPTIONS
  - QUANTIFY PERFORMANCE
  - ESTIMATE RISK

- VERIFICATION PHASE
  - COMPONENT, ASSEMBLY & SUBSYSTEM LEVEL PERFORMANCE
  - SYSTEM LEVEL PERFORMANCE
  - HARDWARE/SOFTWARE INTERFACES

- OPERATIONS PHASE
  - GROWTH
  - TECHNOLOGY IMPROVEMENTS
  - NORMAL MAINTENANCE

Facilities

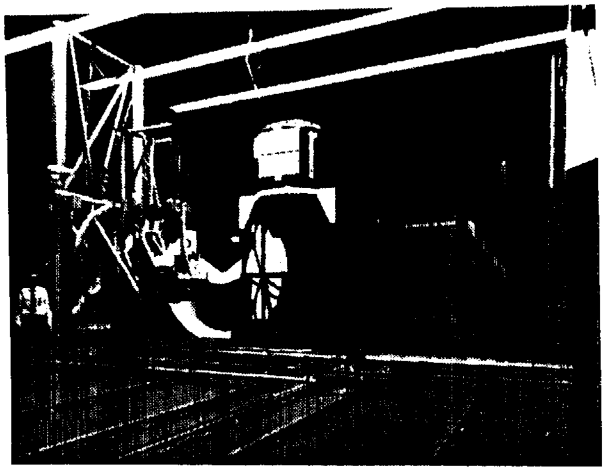
Three facilities at Grumman are applicable to engineering development of proximity operations systems. These include the Large Amplitude Space Simulator (LASS), a manipulator lab and a controls and display lab.

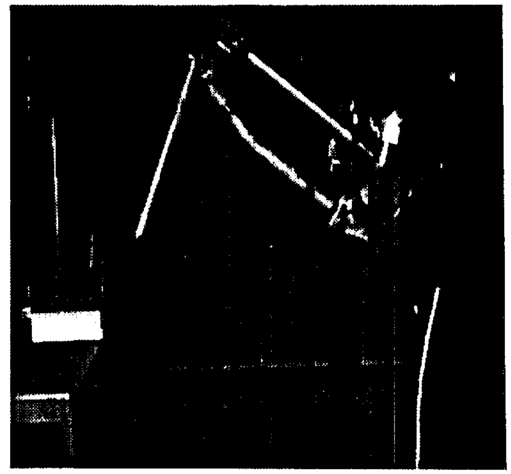
The LASS is a motion base that has been used in the development of the Manipulator Foot Restraint, as well as engineering evaluations of remote free flyers, berthing systems and payload handling systems.

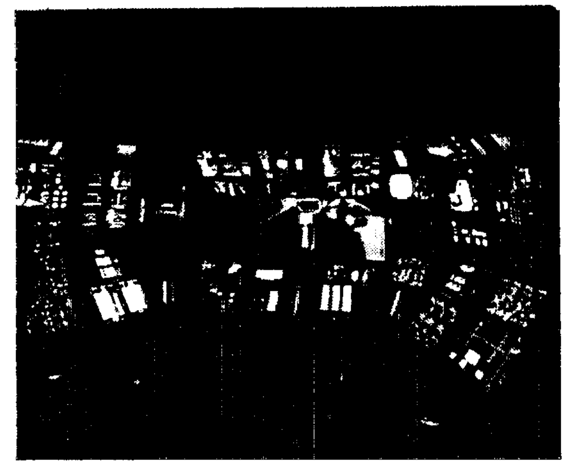
The manipulator laboratory houses a 6 DOF master slave arm for evaluating teleoperations and robotics as well as development of AI and supervisory control techniques as it relates to space station proximity operations.

The controls and displays lab is adjacent to the motion base and houses equipment for evaluating C^3 techniques for proximity operations using man-in-the-loop simulations.

FACILITIES







DEVELOPMENT SYSTEM APPLICATIONS

LARGE AMPLITUTDE SPACE SIMULATOR

- MANIPULATOR FOOT RESTRAINT
- REMOTE FREE FLYER
- BERTHING OPS
- P/L HANDLING SYSTEMS

MANIPULATOR LAB

- TELE OPERATIONS
- ROBOTICS
  - AI
  - SUPERVISORY CONTROLS

CONTROLS & DISPLAYS LAB

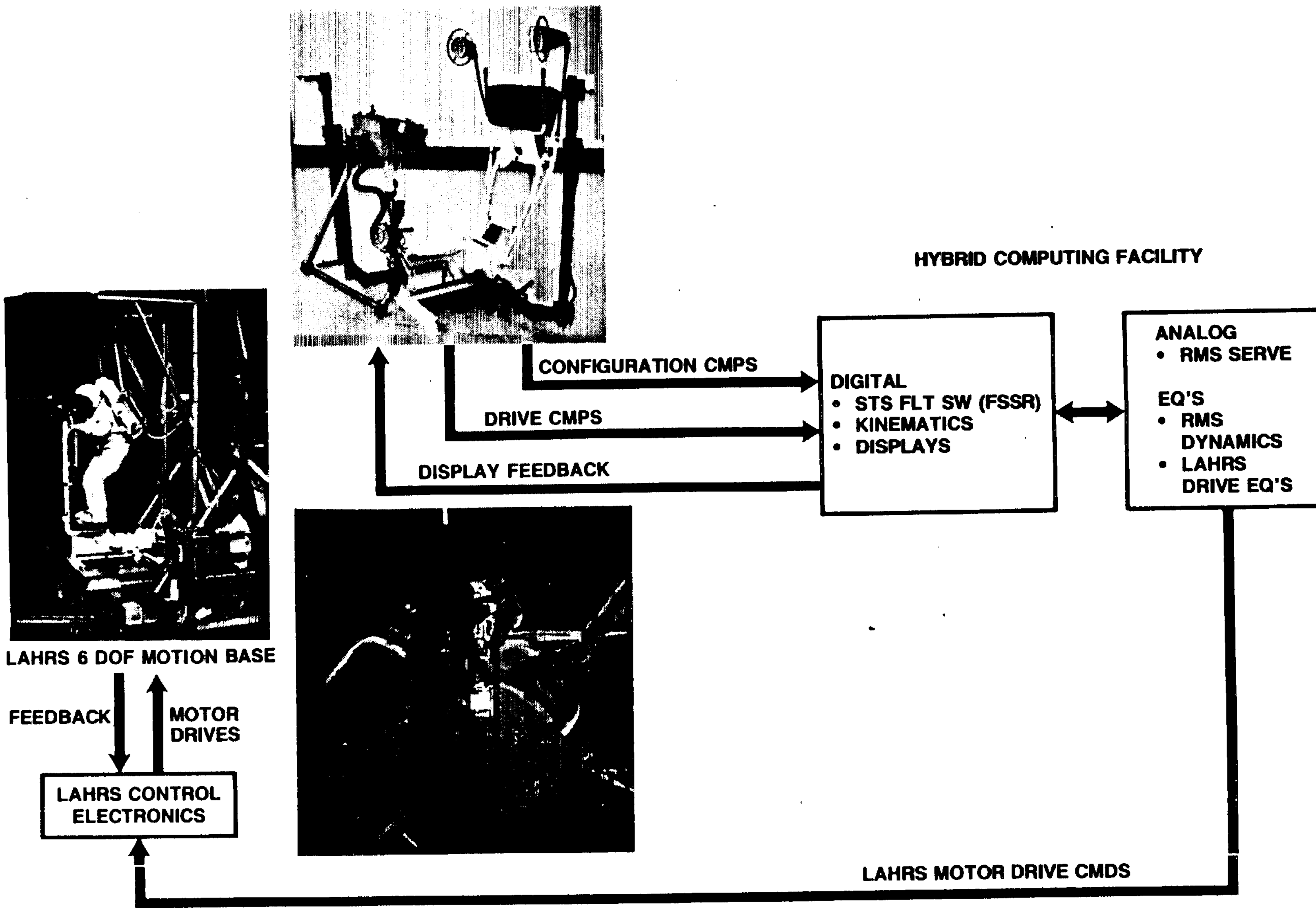
- PROX OPS INTERFACES
- SUBSYSTEM MONITORING INTERFACES
- ADVANCED HARDWARE SYSTEMS EVALUATIONS

Large Amplitude Space Simulator

The LASS is a 6 DOF motion base that is interfaced to a hybrid computer math model to represent the tip motion of the Shuttle's Remote Manipulator System. The digital computer houses the STS flight software, computes arm kinematics and provides the interface with controls and displays. The analog portion of the systems models the joint servos of the RMS.

The motion base has sufficient load carrying capability to lift a full scale test article, such as the Manipulator Foot Restraint, as well as a fully suited test subject and a safety officer.

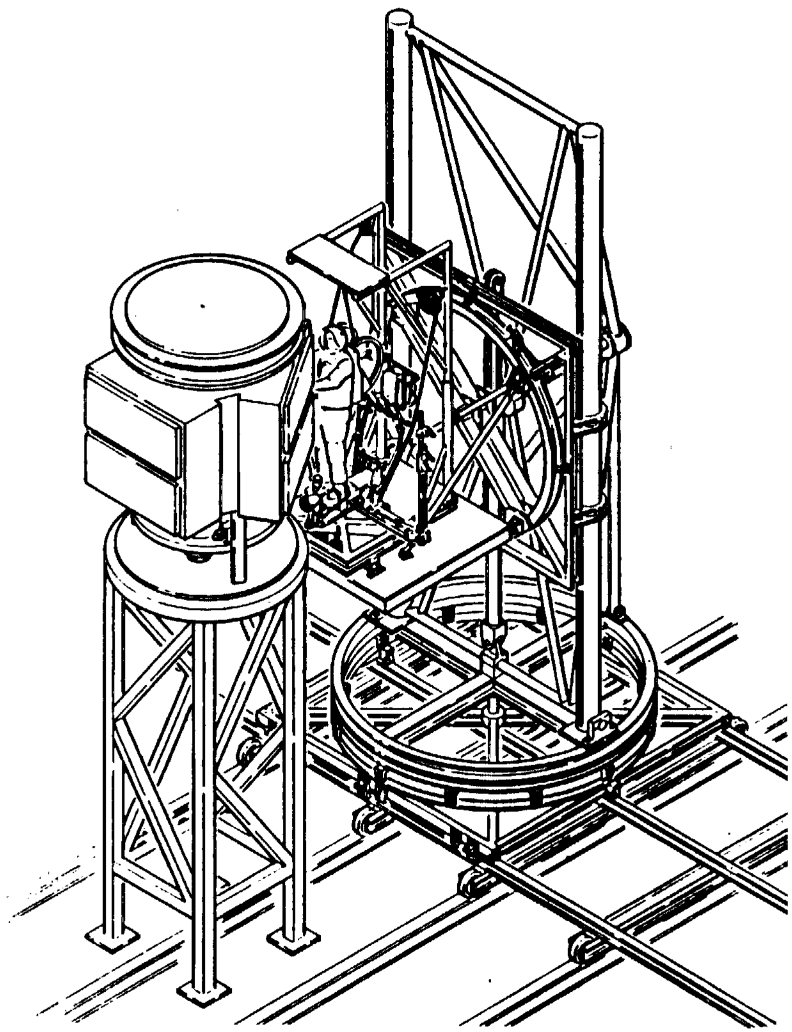
GRUMMAN'S LARGE AMPLITUDE SPACE SIMULATOR (LASS)



LASS Facility Characteristics

The facing page summarizes the characteristics of LASS. The gaming area is 50 ft by 50 ft. with a vertical motion of 9 ft. Pitch and roll are constrained at $\pm 10°$ but could be opened. Yaw motion is continuous.

LASS FACILITY



FACILITY CHARACTERISTICS

MOTION	
VERTICAL	— 9 FT
HORIZONTAL	
X	— 50 FT
Y	— 50 FT
PITCH	— 10°
ROLL	— 10°
YAW	— 360°
VELOCITY	
LINEAR	— 2 FT/SEC
ROTATIONAL	— 6°/SEC
ACCELERATION	
LINEAR	— 6 FT/SEC^2
ROTATIONAL	— 18°/SEC^2

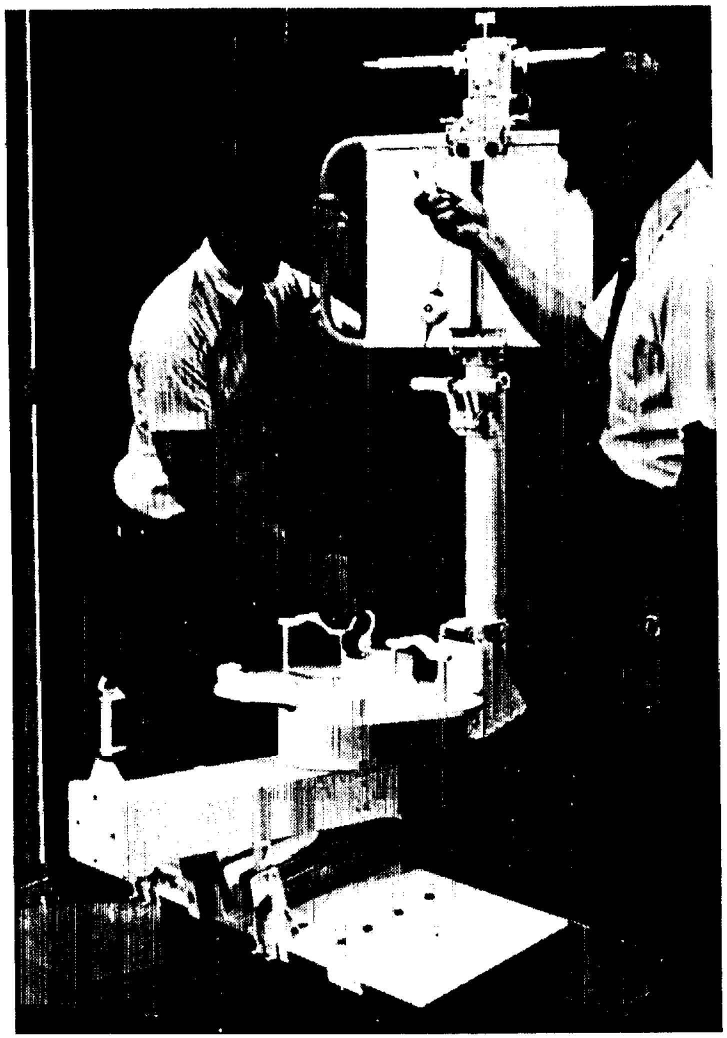
FREQUENCY BANDWIDTH ~ 1Hz

LASS Test Series Objectives

The objectives of the four test series used in the development of the Manipulator Foot Restraint are summarized on the facing page. Test 1 emphasized evaluation of handling qualities, selection of control modes and evaluation of control devices (hand controllers or switches). The second test series evaluated the effect of RMS dynamics on work productivity and evaluated the need and operation of a stabilizer.

Test series 3A and 3B evaluated operations for replacing the Multimission Modular Spacecraft subsystems using a GSFC-supplied high fidelity mock-up, and replacing a LDEF experiment tray. Test Series 3A was performed with shirt-sleeve subjects with emphasis placed on developing procedures. Test Series 3B was a repeat of Test 3A using suited subjects and emphasized the study of work-induced loads on RMS dynamics and a comparison of crew productivity using the MFR alternate configurations.

SIMULATION TESTS LEADING TO MANIPULATOR FOOT RESTRAINT



TEST OBJECTIVES

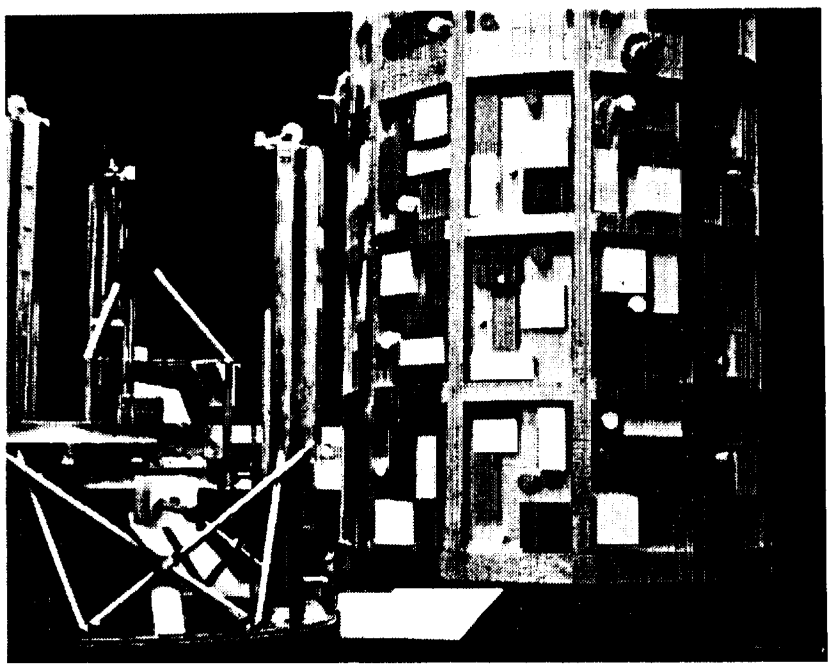
TEST 1	—	EVALUATE OCP HANDLING QUALITIES
	—	COORDINATE REF SYSTEMS CONTROL MODES
	—	CONTROL DEVICES
TEST 2	—	EVALUATE EFFECTS OF RMS STRUCTURAL DYNAMICS ON OPERATOR'S ABILITY TO PERFORM SATELLITE SERVICE TASKS
TEST 3	—	EVALUATE REPLACING MULTIMISSION MODULAR SPACECRAFT S/S MODULES (C&DH AND PROPULSION)
	—	EVALUATE REPLACING LDEF EXPERIMENT TRAYS
TEST 4	—	EVALUATE SERVICING & REPLACING SPACE TELESCOPE EQUIPMENT
	—	EVALUATE REPLACING LANDSAT SCIENTIFIC INSTRUMENTS

ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

Remote Free Flyer Development

The facing page summarizes test objectives for evaluating operations of a Remote Free Flyer. The LASS computer system was modelled to represent a small proximity vehicle operated from the aft flight deck of the Shuttle or from the Space Station. Control modes and docking interfaces have been evaluated and future work is planned to evaluate the capabilities to capture tumbling satellites in the presence of communications time delays.

REMOTE FREE FLYER DEVELOPMENT



TEST OBJECTIVES

- CONTROL MODE SELECTION
- DOCKING TARGET EVALUATIONS
- FEASIBILITY OF TUMBLING SATELLITE CAPTURE
- FEASIBILITY OF OPERATIONS WITH TIME DELAY

ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

Control Mode Comparisons

The facing page is a comparison of the time regarded to perform final Approach and Capture of a satellite for two rotational control modes, Minimum Impulse and Attitude Hold with Acceleration Command. For each control mode three cases of control response were tested. The translational control mode was Acceleration Command for all cases and modes.

CONTROL MODE COMPARISONS

AVERAGE TIME (IN MINUTES) AND VARIABILITY TO PERFORM THE FINAL APPROACH AND CAPTURE OF A SATELLITE AS A FUNCTIN OF ATTITUDE CONTROL MODE

	MINIMUM PULSE					
	0.1 DEG/SEC		0.5 DEG/SEC		1.0 DEG/SEC	
	MEAN	S.D.	MEAN	S.D.	MEAN	S.D.
SUBJECT A (N = 5)	2.81	0.38	2.35	0.24	2.55	0.78
SUBJECT B (N = 5)	3.64	0.25	3.03	0.44	2.81	0.30
GROUPED (N = 5)	3.22	0.54	2.69	0.49	2.68	0.58

	ACCEL CMD WITH ATT HOLD					
	0.7 DEG SEC(2)		1.8 DEG/SEC(2)		3.0 DEG/SEC(2)	
	MEAN	S.D.	MEAN	S.D.	MEAN	S.D.
SUBJECT A (N = 5)	2.24	0.29	2.56	0.36	2.25	0.33
SUBJECT B (N = 5)	3.79	0.50	3.58	1.00	3.83	0.58
GROUPED (N = 10)	3.02	0.90	3.07	0.88	3.04	0.95

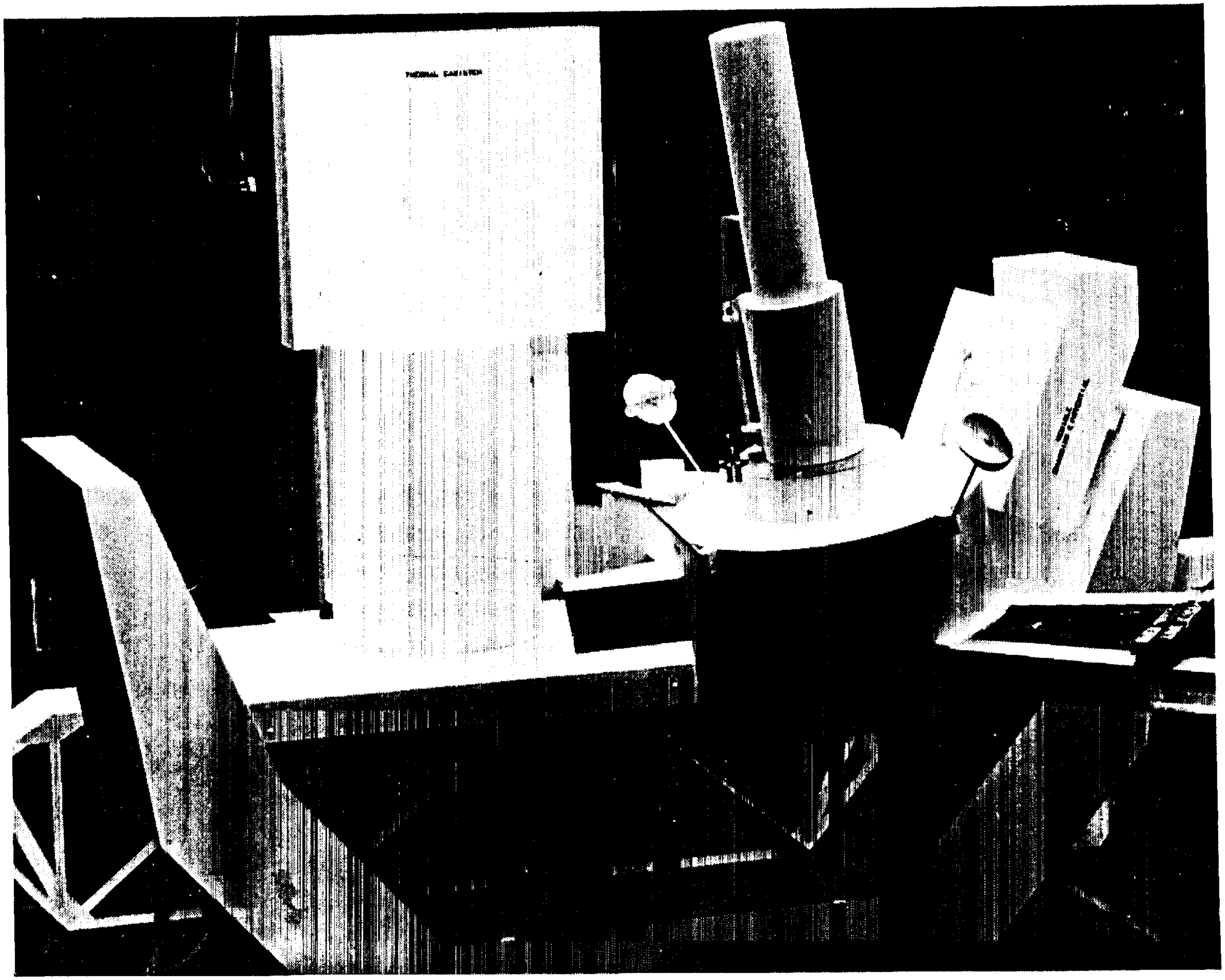
Constrained Motion/Berthing Simulation

The facing page presents a test set-up that evaluated RMS installation of the PDP. To simulate the constrained motion conditions, a force-torque sensor was mounted at the end of the motion base and fed back to the computer model to determine back drive dynamics on the arm.

The results of the simulation were not satisfactory in that the time delays in the system cause instabilities. Proper application of reduced gain on the force torque sensor and increased joint damping stabilized the system. Simulation of the back drive/constrained motion was accomplished but it was not determined if the simulation was valid. To accomplish this, comparative tests between flight and simulator performance would be needed.

The same need to validate the effects of impacts and induce forces is also apparent in simulating berthing dynamics. Technique to successfully perform the simulation can be developed but testing in terms of a comparison with flight data is needed.

CONSTRAINED MOTION/BERTHING SIMULATIONS



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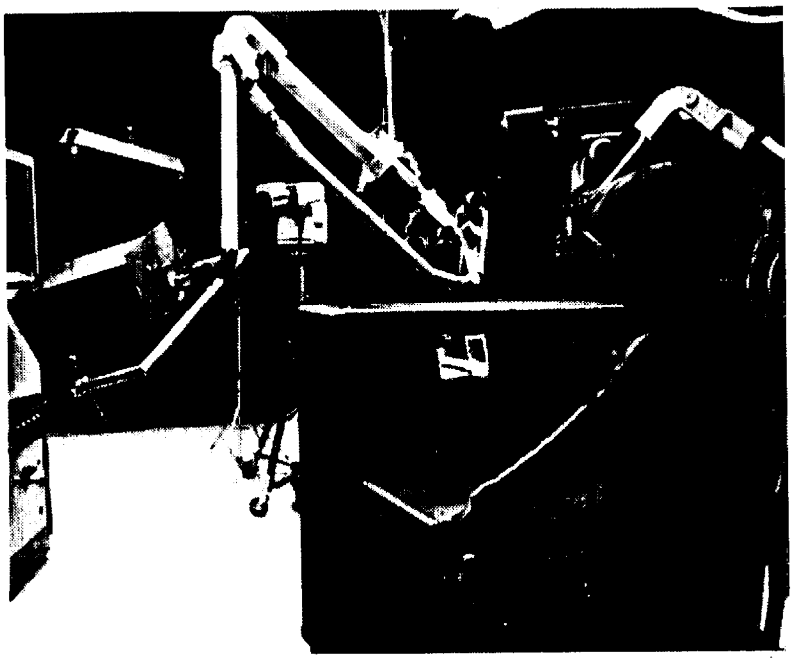
Manipulator Lab

The facing page lists test capabilities available in the Grumman manipulator lab. In addition to a master/slave manipulator that provides force feed back to the operator, TV cameras, 3 DOF hand controllers, lighting system and microprocessor control systems are used to evaluate alternative control modes, man-machine-interfaces and the potential for automated modes of operation.

MANIPULATOR LAB



- MANIPULATOR SYSTEM
  - MASTER/SLAVE
  - RESOLVED RATE
  - HAND CONTROLLER EVALUATION
  - 3DOF/6DOF
- VISION SYSTEMS
  - STEREO/MONO
  - COLOR
  - LIGHTING SYSTEMS
  - RECOGNITION SYSTEMS
- COMPUTER ARCHITECTURE
  - DISTRIBUTED CONTROL OF ARM
  - ROLE OF GRAPHICS
- MISSION SIMULATION
  - TIME DELAY
  - DESIGN FOR SERVICING

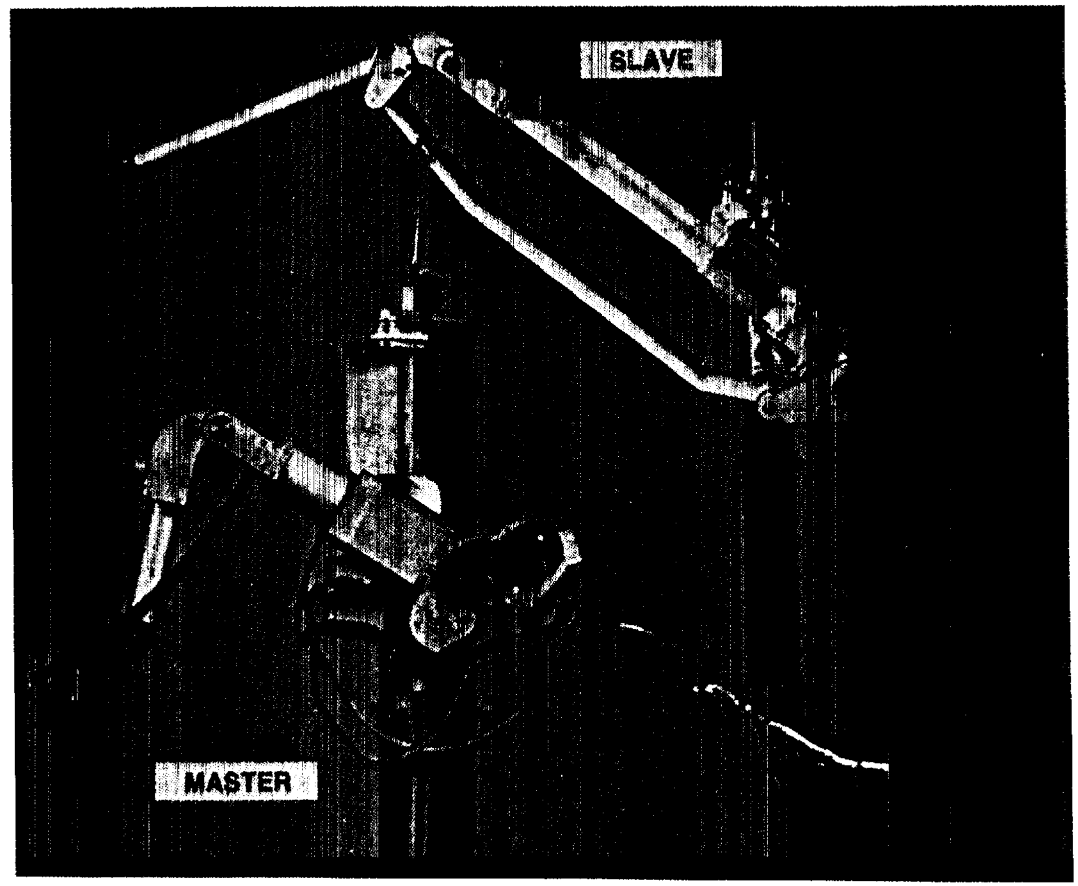
Manipulator Arm Characteristics

The facing page summarizes the characteristics of the master slave system. The master is 1/3 the size of the slave in an effort to minimize the working volume of the controller to be compatible with a small crew cabin. Evaluation of alternate controllers is a prime objective of the facility.

Other features built into the arm is the ability to index the arm in a joint by joint mode and to revert back to the bilateral force mode in a skewed master to slave condition. The arm design also features counter-weights that are repositioned to counterbalance the arm in any orientation.

LABORATORY MANIPULATOR SYSTEM DESCRIPTION

- TYPE: ELECTRO MECHANICAL
- CONTROL: REPLICA MASTER/SLAVE
- FORCE REFLECTION: OPTIONAL
- ARM LENGTH: SLAVE -2m
  MASTER -0.6m
- DOF: 6 + MANUAL YAW + TONG
- WORK VOLUME: 1100 FT^3
- INDEXING OPTION
- VELOCITY RATIO: S/M -3.3:1 & 2:1
- MAX TIP SPEED: SLAVE 30 IN./SEC
- MAX TIP FORCE: SLAVE 15 LB
  MASTER 7 LB
- TRANSMISSION: TENDONS
- COUNTERWEIGHTS: MOTORS
- MANUAL LOCKS & TONG: SLAVE ARM
- ALTERNATE MOUNTINGS



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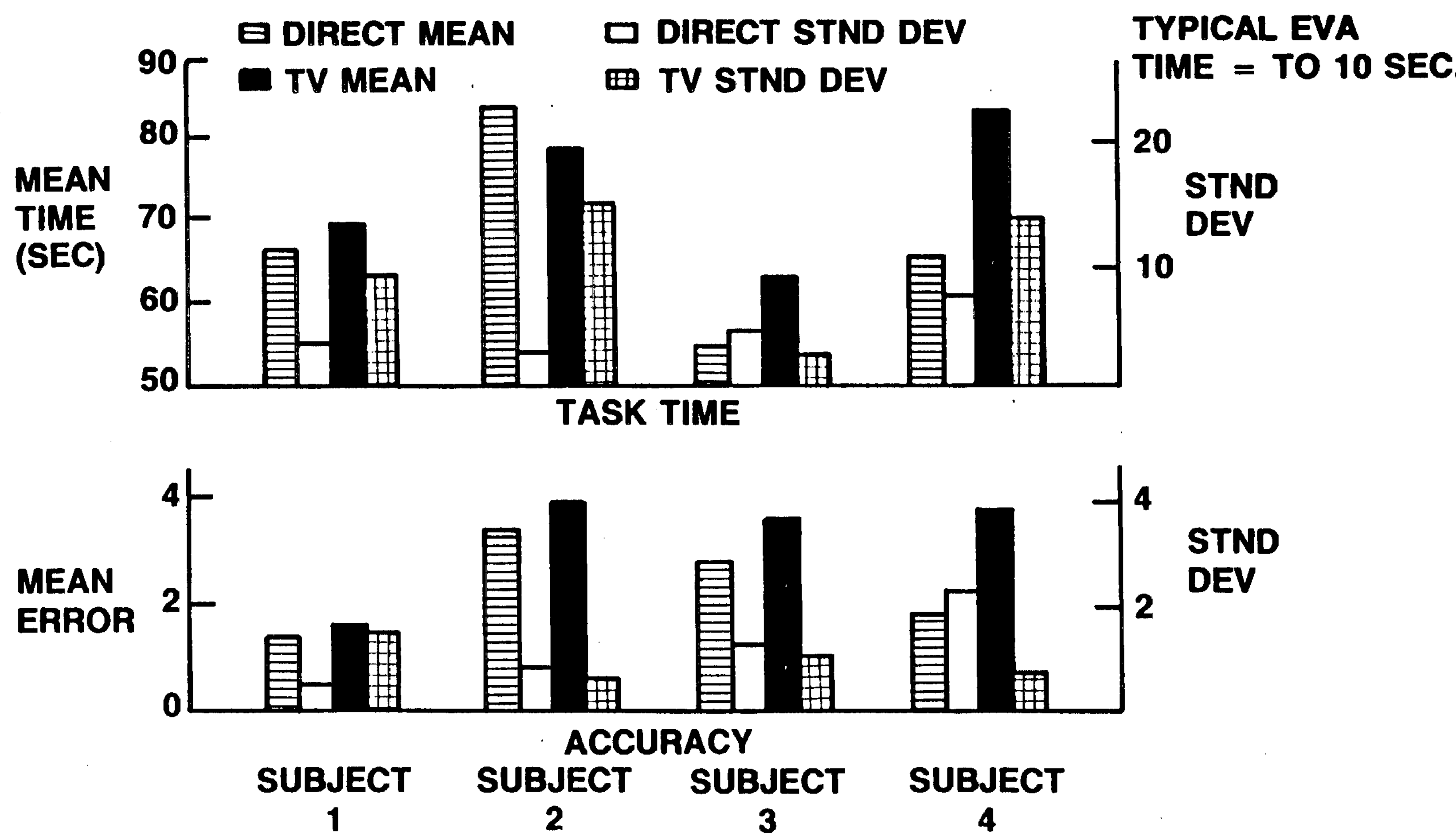
Task Performance Comparisons

In previous simulations, Grumman performed typical space servicing tasks three ways: (1) by a suited astronaut, using direct vision and manual manipulation, (2) by a man using direct vision and a force-feedback manipulator, and (3) by a man working with television and the manipulator. The facing page illustrates the results. The suited astronaut completed the tasks in 5 to 10 sec. The manipulator/direct vision operators took an average of 65 sec to perform the same tasks, with a standard deviation of about 5 sec. The remote teleoperators took an average of 75 sec with a standard deviation of about 11 sec. The mean error for all manipulator activities was about 3. The EVA astronaut is the superior performer, once he gets to the job.

TASK PERFORMANCE COMPARISONS

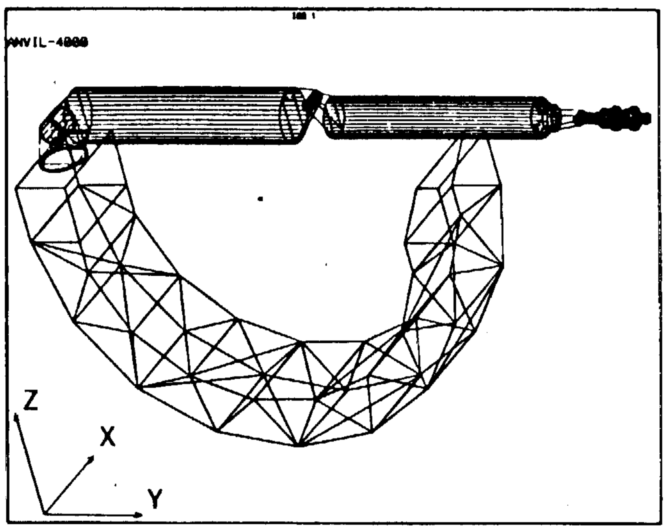
- 3.3 TO 1 POSITION RATIO
- FEEDBACK
- TRIALS 16-20



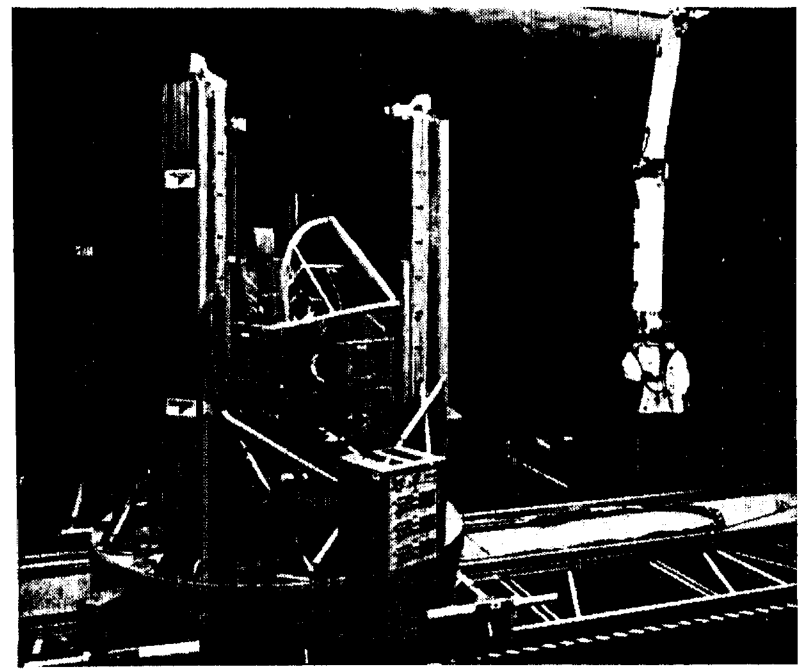
Handling and Positioning Aid (HPA)

The facing page illustrates two simulation approaches that are needed in Proximity Operations development. Graphic systems, as the one illustrated for the HPA, are useful tools for determining geometries and evaluating structure bending. The full scale simulation on the left is useful for evaluating man-machine interfaces.

HANDLING & POSITIONING AID



GRAPHICS SYSTEM
FOR PRELIMINARY
DESIGN

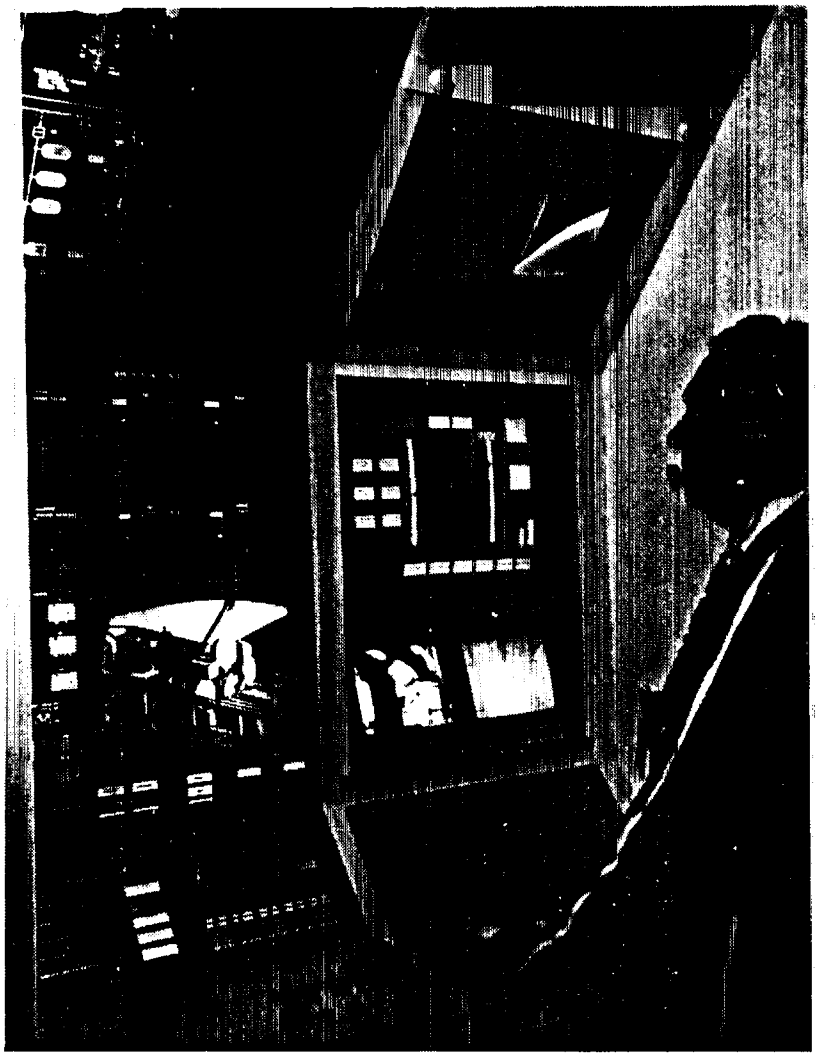


HARDWARE SYSTEM FOR
FINAL VERIFICATION
CONTROL SYSTEM DESIGN
& EVALUATION OF
MANNED MACHINE
INTERFACES

Advanced Controls & Displays Lab

The facing page presents a list of issues that can be addressed with the Controls & Displays Lab. This lab, adjacent to the LASS, is used to evaluate controls and displays with active man-in-the-loop simulation. Concepts such as voice control of CCTV or manipulators are evaluated as are the development of controls for remote free flyers and manipulators.

ADVANCED CONTROLS & DISPLAY LAB



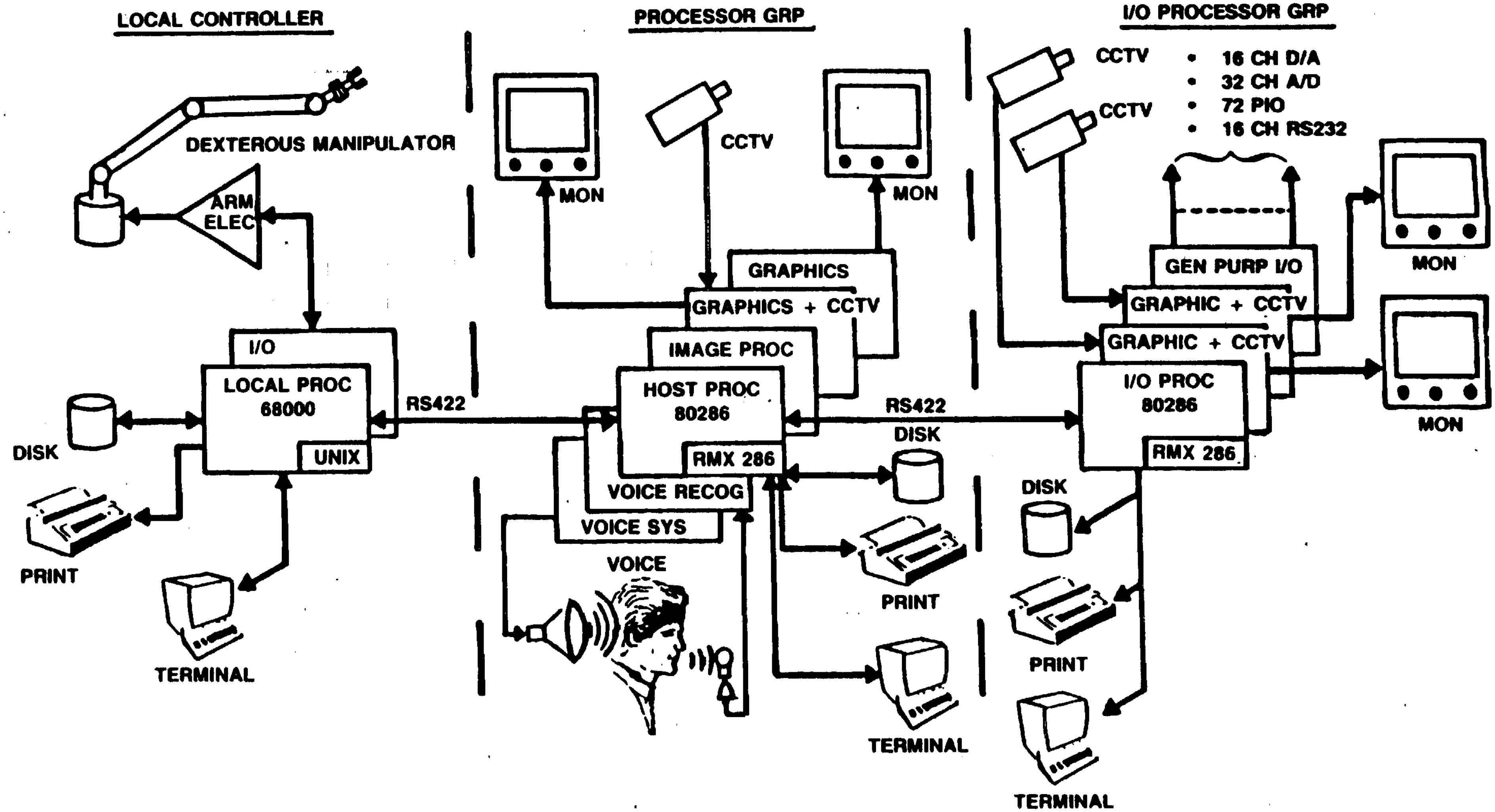
DOCKING DISPLAY WITH
TOUCH PANEL SWITCH
FUNCTIONS

- GRAPHICS EQUIPMENT
  - COLOR VS MONOCHROMATIC
  - FLAT PANEL TECHNOLOGIES
  - HI/LOW RESOLUTION
  - USES OF 3D GRAPHICS
- VOICE RECOGNITION/VOICE SYNTHESIS
- COMPUTER ARCHITECTURE
- INPUT/OUTPUT
  - TOUCH PANEL
  - JOY STICK
  - TRACK BALL
  - REPROGRAMMABLE FUNCTION SWITCHES
- DISPLAY FORMATS
  - PROX OPS
  - SUBSYSTEM MONITOR/CONTROL
  - PAYLOADS

Functional Diagram

The C&D facility utilizes commercial equipment in a network that links local controllers, such as the 68000 driven dexterous manipulator and CCTV's with a processor that generates graphics for display. Included in the system is a capability for voice command/synthesis, and the ability to overlay graphics with CCTV images. This feature is particularly valuable in assessing pro operations display formats, including switch functions on touch panels.

ADVANCED SPACE COCKPIT FUNCTIONAL BLOCK DIAGRAM



Conclusions

- Engineering simulators play an important part in the man-machine design of Prox Operations systems.
- Because of the phasing of Prox OPS Development, development simulators should be planned throughout the life of Space Station.
- Simulation technology issues that should be supported:
  - Effective off-load systems to simulate the dexterity man has in an EVA Environment
  - Development and validate techniques for simulating impact dynamics.

RENDEZVOUS AND DOCKING TECHNOLOGY DEVELOPMENT FOR FUTURE EUROPEAN MISSIONS

BY

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RENDEZVOUS AND PROXIMITY OPERATIONS WORKSHOP

LYNDON B. JOHNSON SPACE CENTER

HOUSTON, TEXAS

19 - 22 FEBRUARY 1985

8-139

esa

RENDEZVOUS AND DOCKING

MAIN ISSUES OF EUROPEAN DEVELOPMENT PROGRAMME FOR RVD

DRIVERS:	- WHY INTEREST IN RENDEZVOUS AND DOCKING ? - WHY EMPHASIS ON UNMANNED RVD ? - WHAT ARE THE POTENTIAL DRIVER MISSIONS ? - WHAT ARE THE NATURAL AND TECHNICAL CONSTRAINTS ?
FUNCTION:	- HOW DOES THE AUTONOMOUS RVD PROCESS FUNCTION ? - WHAT HARDWARE AND SOFTWARE FUNCTIONS ARE REQUIRED ? - WHAT OPERATIONS TECHNIQUES AND CONCEPTS ARE TO BE DEVELOPED ? - HOW CAN MISSION SAFETY BE ASSURED ?
DEVELOPMENT PROGRAMME:	- WHAT DOES ESA WANT TO ACHIEVE IN THE RVD TECHNOLOGY DEVELOPMENT PROGRAMME ? - WHAT ITEMS HAVE TO BE DEVELOPED ? - HOW DO THOSE DEVELOPMENTS FIT TOGETHER ?
IN-ORBIT VERIFICATION & DEMONSTRATION	- WHAT RVD ELEMENTS CAN BE VERIFIED ON HOST MISSIONS ? - WHAT CAN BE VERIFIED ONLY IN A DEDICATED DEMONSTRATION MISSION ?
STANDARDISATION:	- WHAT INTERFACES FOR UNMANNED RVD NEED TO BE STANDARDIZED BETWEEN NASA AND ESA ?

RENDEZVOUS AND DOCKING

WHY INTEREST IN RENDEZVOUS AND DOCKING (RVD)?

RVD IS A PREREQUISITE FOR ADVANCED SPACE OPERATIONS SUCH AS:

- ASSEMBLY IN SPACE
- RETRIEVAL
- SERVICING
- MAINTENANCE AND REPAIR

FOR CASES OF

- SPACECRAFT EXCEEDING THE LAUNCH CAPABILITY OF A SINGLE LAUNCHER
- MODULES TO BE LAUNCHED SEPERATELY FOR ADD-ON OR EXCHANGE
- EXCHANGE OF EXPERIMENTS, MATERIAL SAMPLES ETC.
- REFUELLING, REPLENISHMENT AND SCHEDULED MAINTAINANCE
- RESCUE MISSIONS

FOR THESE CASES AN UNMANNED RVD CAPABILITY IS REQUIRED FOR EUROPE

RENDEZVOUS AND DOCKING

WHY EMPHASIS ON UNMANNED RVD?

- INTERVENTION CAPABILITY BY MAN HIGHLY DESIRABLE IN BUILD UP PHASE
  - UNEQUALLED VERSATILITY AND PROBLEM SOLVING CAPABILITY OF MAN

HOWEVER,

- WHEN NEW OPERATIONS BECOME ROUTINE MATTERS INVOLVEMENT OF HUMAN OPERATORS MAY BECOME UNECONOMIC
  - COST PENALTY OF TRANSPORTING MAN AND LIFE SUPPORT EQUIPMENT INTO ORBIT
- INVOLVEMENT OF ASTRONAUTS MAY BE EXCLUDED A PRIORI
  - EXCESSIVE DISTANCES
  - LONG MISSION DURATIONS
  - HOSTILE ENVIRONMENT

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RENDEZVOUS AND DOCKING

WHY EMPHASIS ON UNMANNED RVD?

RVD OF MANNED SPACECRAFT WILL EVENTUALLY ALSO BE REQUIRED FOR EUROPE

- WITHIN SPACE STATION SCENARIO
- FOR FUTURE AUTONOMOUS EUROPEAN MANNED VEHICLES

HOWEVER,

EUROPE CANNOT PRESENTLY CONTRIBUTE SUBSTANTIALLY TO THE DEVELOPMENT OF MANNED RVD

- US LEAD OF MORE THAN 2 DECADES
- BECAUSE OF SAFETY CRITICALITY RVD WITH THE SPACE STATION ITSELF WILL REMAIN RESPONSIBILITY OF NASA
- RVD INTERFACES DEVELOPED BY NASA WILL BE IMPOSED AS THE STANDARD FOR COMPATIBILITY AND SAFETY REASONS
- THERE WILL BE A STRONG DRIVER FOR FUTURE EUROPEAN MANNED VEHICLES TO USE THE SAME RVD INTERFACE FOR OVERALL MISSION AND RESCUE COMPATIBILITY

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RENDEZVOUS AND DOCKING

TWO POTENTIAL DRIVER MISSIONS

SERVICING OF A MATERIAL PROCESSING PLATFORM

- RVD IN LEO WITHOUT RELAY SATELLITE
  - SPORADIC VISIBILITY
  - HIGH ON-BOARD AUTONOMY
  - SUCCES VERIFICATION BY GROUND
- RVD IN LEO WITH RELAY SATELLITE
  - $\frac{1}{2}$ ORBIT VISIBILITY
  - LOWER ON-BOARD AUTONOMY
  - INITIATION, SEQUENCING MAINLY BY GROUND

ASSEMBLY OF A TELECOMMUNICATION PLATFORM

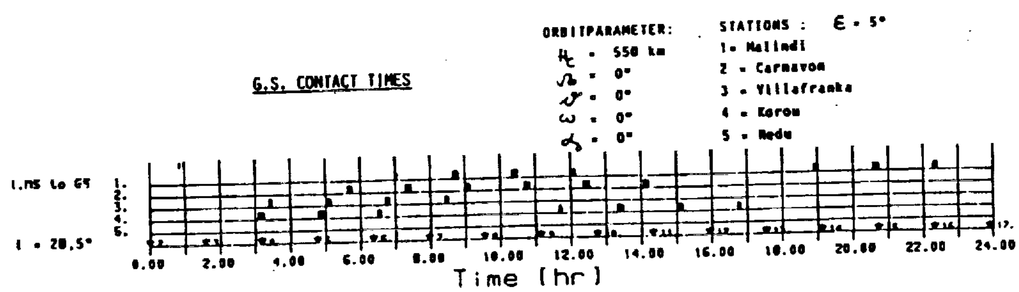
- RVD IN GEO
  - PERMANENT VISIBILITY
  - LIMITED ON-BOARD AUTONOMY
  - INITIATION AND CONTROL MAINLY BY GROUND
- RVD IN GTO
  - LARGE VISIBILITY PERIODS, BUT DISCARDED BECAUSE OF LAUNCH CONSTRAINTS

WHAT EXTERNAL FACTORS AFFECT THE RVD CONCEPT?

- ORBITAL FACTORS:
  - ORBITAL PERIOD
  - ORBITAL DISTURBANCES
  - GROUND VISIBILITY WINDOWS
  - LIGHTING CONDITIONS
- TECHNICAL RESOURCES:
  - GUIDANCE AND CONTROL CAPABILITIES OF GROUND CONTROL CENTRE
  - AVAILABILITY OF A RELAY SATELLITE
  - NUMBER AND DISTRIBUTION OF GROUND STATIONS
  - CONSTRAINTS OF SPACE AND GROUND COMMUNICATIONS LINKS

esa RENDEZVOUS AND DOCKING

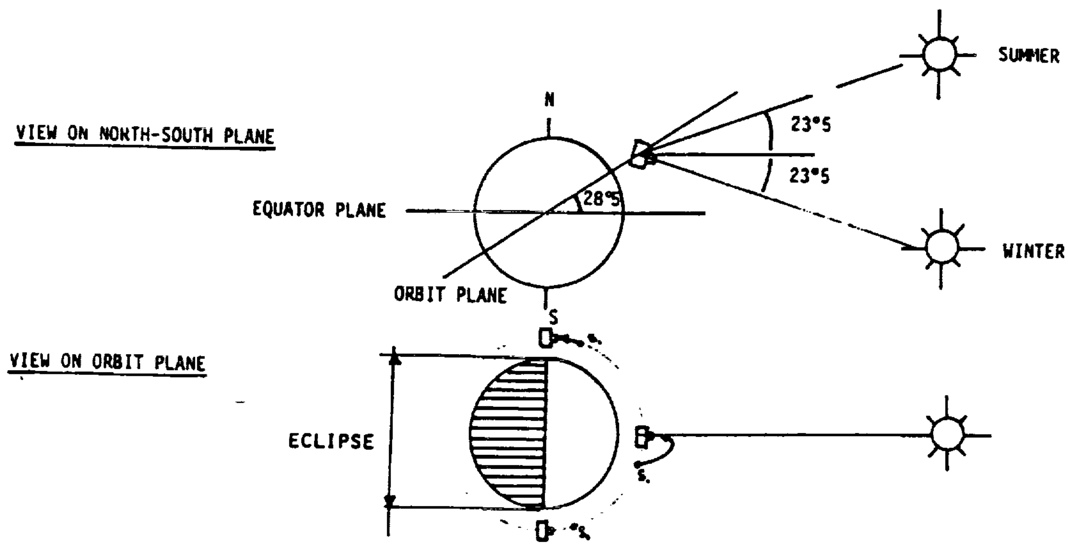
VISIBILITY WINDOWS FOR ESA GROUND STATIONS IN 500km LEO FOR 28.5 deg. (KSC) AND 5 deg. (KOUROU) ORBITS



28.5 deg. ORBIT (KSC LAUNCH)

5 deg. ORBIT (KOUROU LAUNCH)

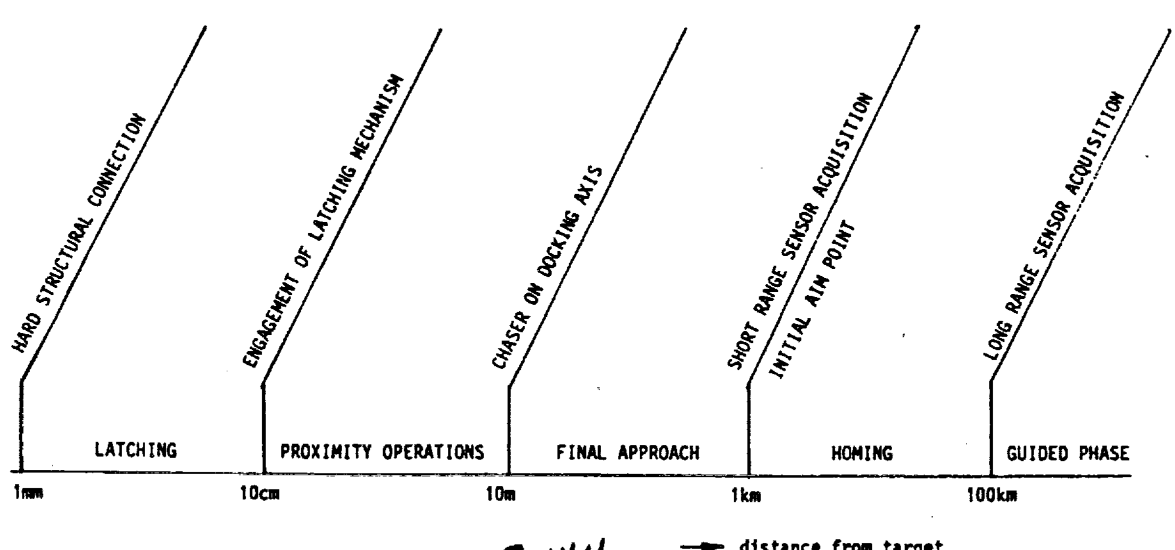
RENDEZVOUS AND DOCKING



ILLUMINATION CONDITIONS AND FLY-AROUND REQUIREMENTS FOR THE MODEL MISSION (example : particular set of orbit conditions)

RENDEZVOUS AND DOCKING

RVD PHASES



RENDEZVOUS AND DOCKING

OPERATIONS CONCEPT

MODEL MISSION:
- RVD IN 500 KM 28,5° LEO WITHOUT RELAY SATELLITE
- ALL ESA GROUND STATIONS AVAILABLE
- CHASER ACTIVE, TARGET SUNPOINTING, PASSIVE

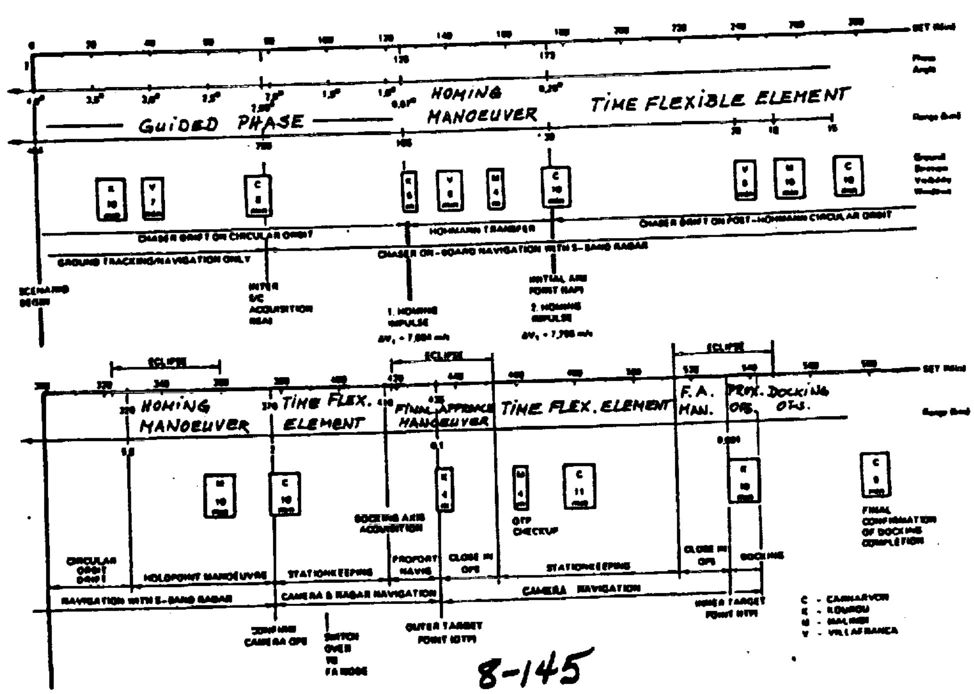
BASIC PHILOSOPHY:
- ALL CONTROL MODES AND MANOEUVRES PERFORMED AUTOMATICALLY ON-BOARD
- AFTER RVD SENSOR ACQUISITION, GROUND INTERVENTION ONLY FOR
  - MONITORING OF PROPER OPERATION, HEALTH CHECKS
  - VERIFICATION OF TRAJECTORY KEY POINTS, UPDATING NAVIGATION
  - INITIATION OF FINAL APPROACH AND TERMINAL CLOSURE PHASES

RVD PHASE SEQUENCING:
- INITIATION OF SUBSEQUENT PHASES ONLY AFTER VERIFICATION OF PREVIOUS ONE
- INTRODUCTION OF INTERMEDIATE TARGET POINTS FOR VERIFICATION

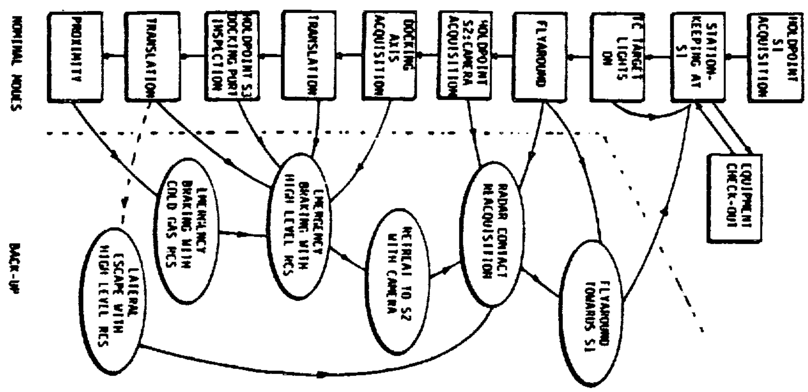
SYNCHRONIZATION PROBLEM:
- RVD PROCESS NOT SYNCHRONIZED WITH GROUND VISIBILITY WINDOWS AND WITH LIGHTING CONDITIONS → INTRODUCTION OF TIME FLEXIBLE ELEMENTS:
  - SLOW DRIFTS
  - PASSIVE HOLD POINTS ON TARGET ORBIT
  - ACTIVE STATION KEEPING

RENDEZVOUS AND DOCKING

TIMELINE OF NOMINAL MISSION



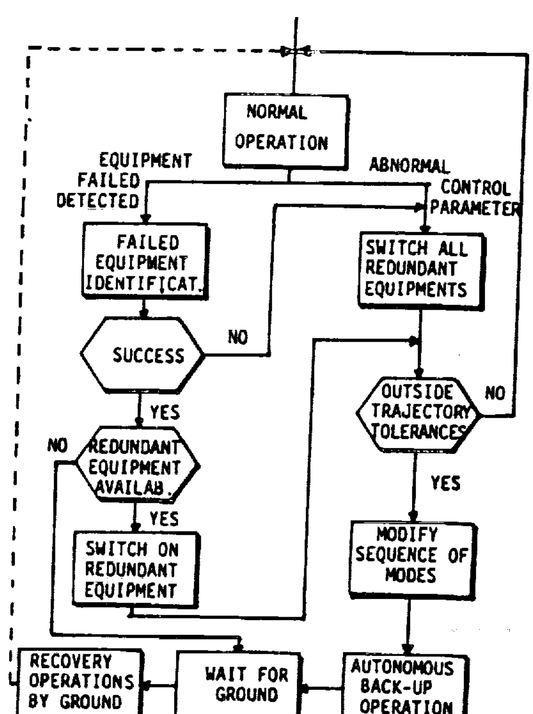
RENDEZVOUS AND DOCKING



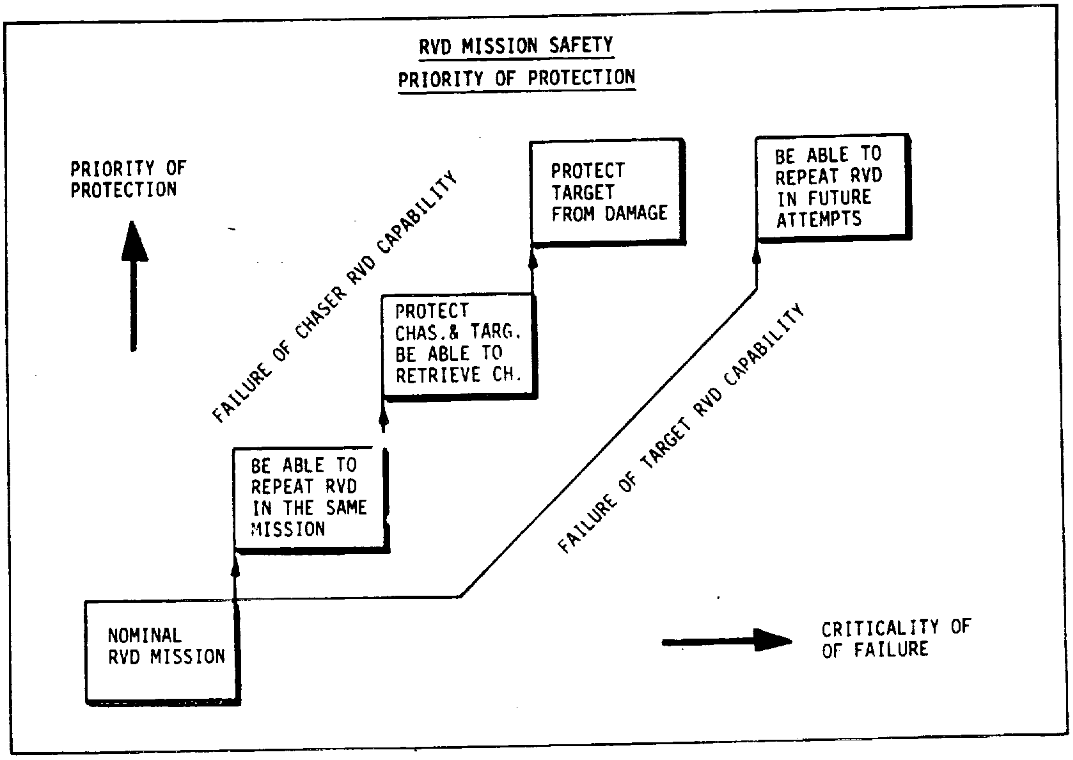
FINAL APPROACH PHASE LOGIC

RENDEZVOUS AND DOCKING

BACK-UP MODES PHILOSOPHY



RENDEZVOUS AND DOCKING



RENDEZVOUS AND DOCKING

WHAT DOES ESA WANT TO ACHIEVE WITHIN THE TECHNOLOGY PROGRAMME ?

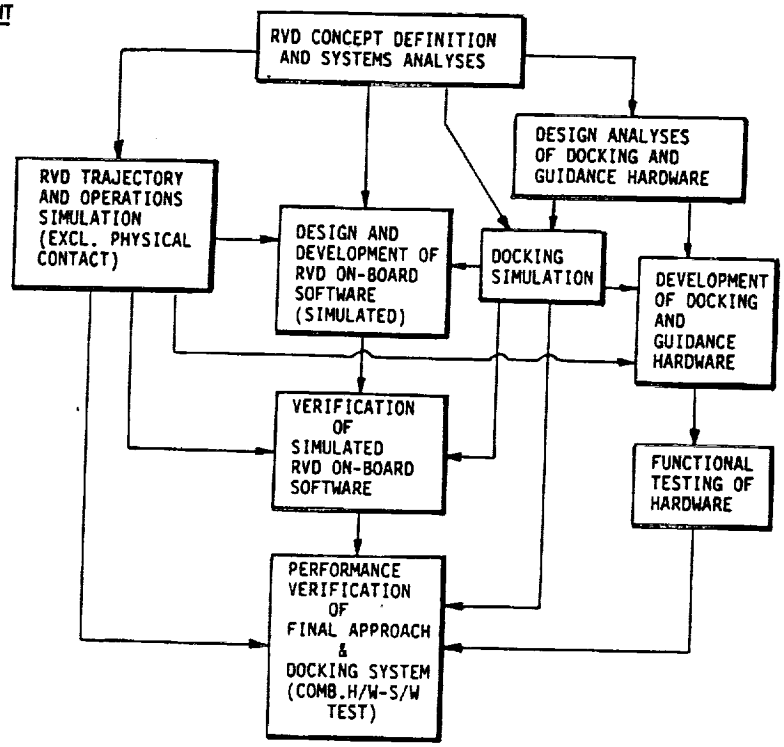
→ TO DEVELOP AND DEMONSTRATE A FUNCTIONAL RVD SYSTEM "BREADBOARD" LEVEL

HOW ? → BY :

- DEVELOPING SUITABLE RVD SYSTEMS AND OPERATIONAL CONCEPTS FOR THE POTENTIAL DRIVER MISSIONS
- ESTABLISHING RVD SIMULATION PROGRAMS FOR GUIDANCE/TRAJECTORY CONTROL AND DOCKING DYNAMICS
- DEVELOPING DOCKING MECHANISM AND GUIDANCE SENSOR HARDWARE
- DEVELOPING AN ON-BOARD SOFTWARE PROTOTYPE FOR A SPECIFIC MODEL MISSION CASE
- DEMONSTRATING THE PERFORMANCE OF THE RVD SYSTEM FOR THE CRITICAL LAST METERS ON A DYNAMIC MOTION SIMULATOR ACCOMODATING SENSORS, DOCKING MECHANISM AND ON-BOARD SOFTWARE

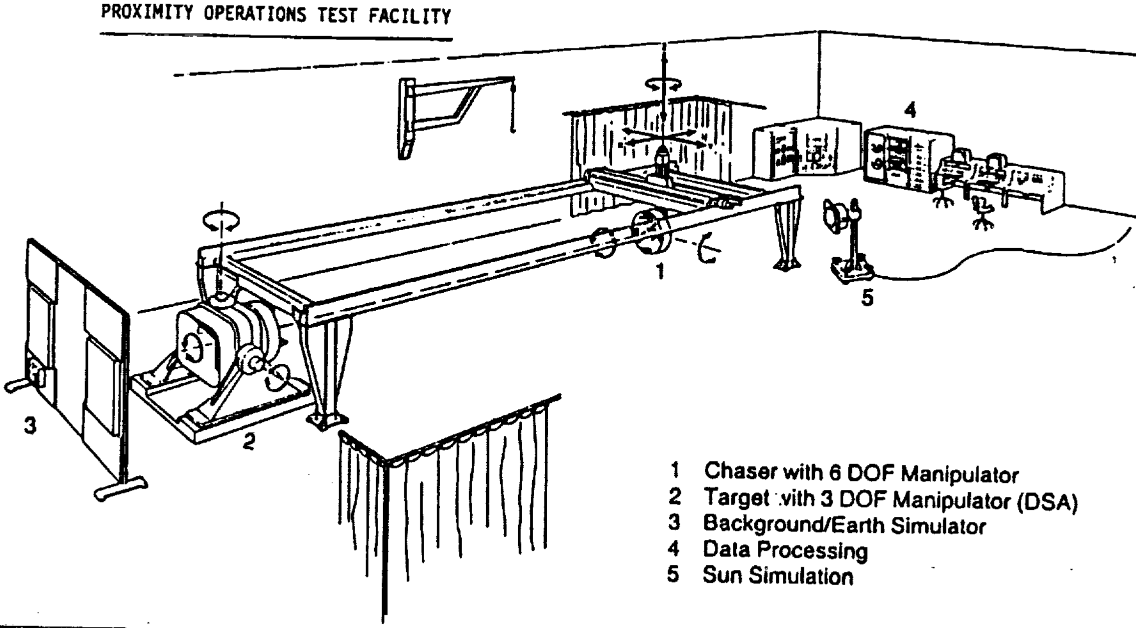
RENDEZVOUS AND DOCKING

LOGIC OF RVD DEVELOPMENT PROGRAMME



RENDEZVOUS AND DOCKING

PROXIMITY OPERATIONS TEST FACILITY



1 Chaser with 6 DOF Manipulator
2 Target with 3 DOF Manipulator (DSA)
3 Background/Earth Simulator
4 Data Processing
5 Sun Simulation

RENDEZVOUS AND DOCKING

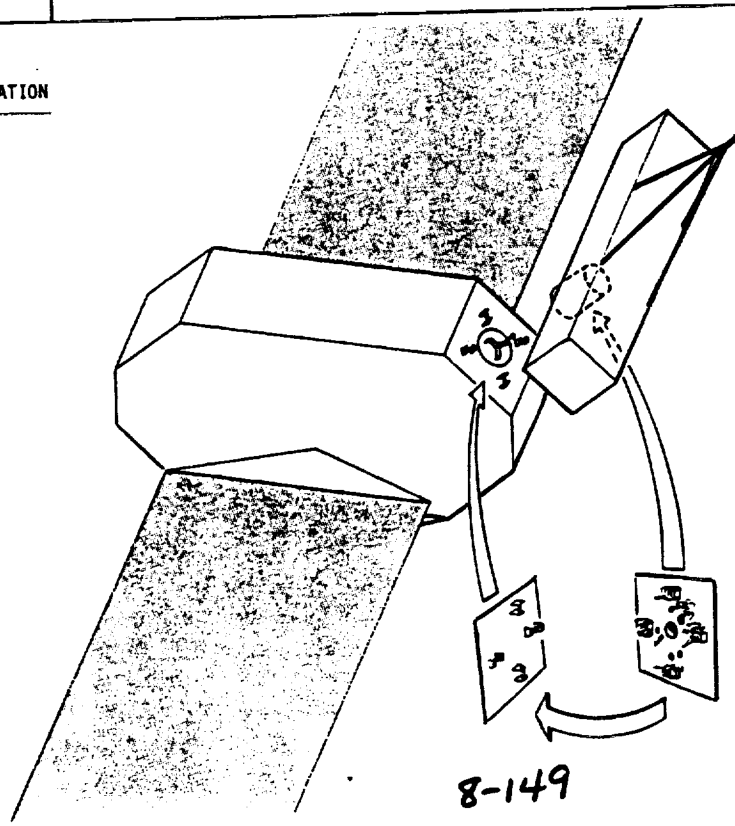
IN-ORBIT VERIFICATION & DEMONSTRATION

- O WHY IN-ORBIT DEMONSTRATION?
  - AUTOMATIC RVD CANNOT BE CONSIDERED READY FOR OPERATIONAL MISSION UNLESS DEMONSTRATED ONCE
  - TRUE VERIFICATION OF THE COMPLETE SYSTEM CAN ONLY BE PROVIDED IN DEDICATED RVD FLIGHT
- O WHAT HAS TO BE DONE PRIOR TO AN RVD DEMONSTRATION MISSION?
  - RVD DEMO MISSION PUTS A RISK INVESTMENT IN FABRICATION AND LAUNCH OF 2 SPACECRAFT
  - RVD SYSTEM MUST BE SUFFICIENTLY VERIFIED PRIOR TO DEMO MISSION TO JUSTIFY RISK
- O HARMONIZED VERIFICATION APPROACH TO BE ESTABLISHED:
  - WHAT CAN BE VERIFIED ON GROUND?
  - WHAT CAN BE VERIFIED IN-ORBIT ON HOST MISSIONS?
  - WHAT CAN BE VERIFIED ONLY BY A FULL RVD DEMONSTRATION MISSION
- O IN-ORBIT DEMONSTRATION OF RVD ELEMENTS PRESENTLY UNDER INVESTIGATION BY THE AGENCY
- O STUDY PROGRAMME FOR RVD DEMO MISSION INVOLVING EURECA AND SECOND EUROPEAN SPACECRAFT
- O DEMONSTRATION OF MUTUAL RVD BETWEEN EUROPEAN AND AMERIACAN VEHICLE WOULD BE HIGHLY DESIRABLE

RENDEZVOUS AND DOCKING

RVD DEMONSTRATION

WITH EURECA



RENDEZVOUS AND DOCKING

STANDARDIZATION OF INTERFACES

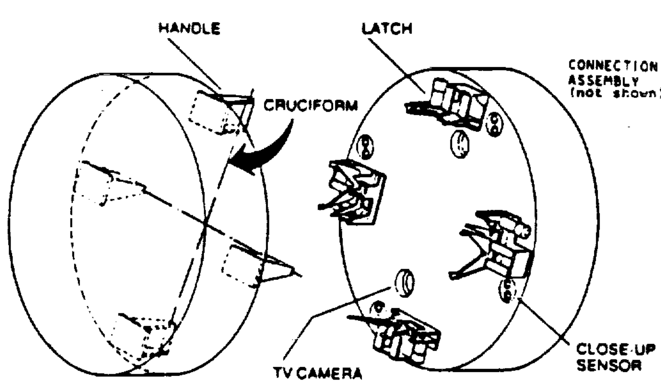
O STANDARDIZATION OF INTERFACES FOR UNMANNED RVD HIGHLY DESIRABLE IN ORDER TO ACHIEVE MISSION COMPATIBILITY

O RVD INTERFACES MUST BE DESIGNED TO MINIMIZE MASS AND VOLUME PENALTIES ON FUTURE MISSIONS

O AS A MINIMUM, INTERFACES FOR FINAL APPROACH SENSORS AND FOR DOCKING/BERTHING SHOULD BE STANDARDIZED

O ESA PROPOSES 4 HANDLES ARRANGED IN CRUCIFORM AS STANDARD DOCKING INTERFACE (THIS IS SIMILAR TO THE BERTHING INTERFACE ON THE SPACE TELESCOPE)

RENDEZVOUS AND DOCKING



AOCS–CLOSURE CONTROLLED

esa

RENDEZVOUS AND DOCKING

CONCLUSIONS :

- THE EUROPEAN SPACE AGENCY (ESA) HAS MADE RVD ONE OF ITS MAJOR TECHNOLOGY THEMES
- MATERIALS PROCESSING PLATFORMS IN LEO AND COMMUNICATIONS PLATFORMS IN GEO ARE DRIVER MISSIONS
- THE RVD CONCEPT PURSUED IS BASED ON A HIGH LEVEL OF AUTONOMY ONBOARD, GROUND INTERVENTION ONLY FOR MONITORING AND HIGHEST LEVEL DECISION MAKING
- DEVELOPMENT FOCUSSED ON MODEL MISSION IN LEO (SPACE STATION SCENARIO)
- DESIGN/DEVELOPMENT OF SYSTEMS & OPERATIONS CONCEPTS AND OF H/W & S/W STARTED COMPLETION BEFORE 1990 PLANNED
- IN-ORBIT DEMONSTRATION OF ELEMENTS PLANNED, FULL DEMONSTRATION MISSION SEEN AS FINAL STEP (JOINT MISSION WITH NASA ?)
- EARLY AGREEMENT BETWEEN NASA AND ESA ON STANDARD SENSOR AND DOCKING INTERFACES FOR UNMANNED RVD NECESSARY
- DETAILS OF EUROPEAN RVD DEVELOPMENT AND DEMONSTRATION PROGRAMME PRESENTED BY P.NATENBRUK (SESSION 8), B.KUNKEL (SESSION 6) AND B.CLAUDINON (SESSION 9)

DEMONSTRATION MISSION FOR AUTONOMOUS RENDEZVOUS WITH THE EUREKA PLATFORM

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MATRA ESPACE

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MBB/ERNO

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AEROSPATIALE

RENDEZVOUS AND PROXIMITY OPERATIONS WORKSHOP
LYNDON B. JOHNSON SPACE CENTER
HOUSTON, TEXAS

19 – 22 FEBRUARY 1985

ABSTRACT

Autonomous Rendezvous and Docking (RVD) in orbit is a new operation requiring in-flight verification prior to operational application. Based on a prescribed mission model in low earth orbit, verification objectives are derived. Alternative flight modes for verification purposes, which may be structured into a continuous, step-by-step RVD qualification program, are identified. The latter terminates with a final rehearsal in form of a dedicated, the whole RVD process covering demonstration mission. Possible involved spacecraft are identified and Eureca platform, as prime contender as target spacecraft, presented. Finally, demonstration mission profile and monitoring concepts are discussed.

REFERENCES

All References relate to papers given in the NASA Rendezvous and Proximity Operations Workshop JSC, 19 – 22 February 1985.

(1) W. Fehse/ESA
Rendezvous Technology Development for Future European Mission Scenarios
Session 8, presentation 7

(2) B. Claudinon / MATRA
Simulation of the Last Meters of Approach and Docking
Session 9, presentation 6

(3) B. Kunkel / MBB–ERNO
Rendezvous and Proximity Sensor Candidates
Session 6, presentation 9

(4) E. Graf / ESA
Rendezvous and Proximity Operations During EURECA Missions
Session 2, presentation 1

CONTENTS

1. INTRODUCTION

The process of Rendezvous and Docking (RVD) performed in an autonomous or automated mode, that is with no involvement of man in the guidance, navigation and control (GN & C) loop, has been the subject of numerous in-depth studies, sponsored by the European Space Agency (ESA) during the last few years. Within these studies, this RVD process was incorporated into diverse scenarios, such as in low earth orbit (LEO), on transfer to geosynchronous height (GTO) or on geostationary orbit (GEO) itself.

Due to the oncoming Space Station program and Europe's interest to participate, the LEO case appears at present as the more probable and immediate one. It is this case where the investigations have been carried far enough to identify the necessary technologies, to assess the maturity of available technologies and to embark upon a technology development program. Accordingly, the planning of such a program has been taken up by ESA (Ref. 1).

The subject RVD process is a new space operation, at least for Europe. Thus, the overall RVD concept as well as the various technologies and operational techniques require comprehensive verification prior to a first application in an operational mission. Basic verification of individual pieces of equipment, of subsystem or of some interacting operations of hardware packages can be provided on ground. Partial qualification of equipment, functions and operations can also be provided in orbital flight not dedicated to RVD such as of single spacecraft. But the mastering of the complete process, over its full range, fully representative of later needs, can probably be demonstrated only in a dedicated RVD-test-flight, termed hereafter as the RVD-Demonstration-Mission.

A brief presentation of one such possible mission, from the European point-of-view and as formulated for the present on the basis of recent studies, is attempted in this paper. The present frame does not allow to elaborate the many considerations associated, but a few key factors affecting mission structure are discussed in the following sections.

2. THE MISSION MODEL

The overall demonstration goal is to successfully perform an RVD flight which is representative of the RVD process, which is foreseen within a future operational mission. Thus, RVD elements to be included in the verification flight depend above all on the type and scope of the latter. Its type and altitude of orbit, its objectives, operational constraints and spacecraft characteristics, impact the associated rendezvous strategies, guidance modes, proximity operations, used hardware, required thrust levels for manoeuvring, illumination conditions, post docking activities or ground station contact conditions and many more.

Taking, as an example, ground station performance in orbit determination and prediction : For a given station, accuracies will depend on contact opportunities, which in turn are functions of orbital height. Orbit determination accuracies, however, determine the least inter-spacecraft

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acquisition-range at which autonomous RVD may be initiated. This, in turn, dictates ranging requirements posed to the onboard long-range navigational sensor and thus affects the associated sensor qualification objectives as well as the related demonstration of flight modes prior to and following the inter-spacecraft-acquisition.

In sum, full understanding of the operational mission envisaged is necessary in order to define and tailor to it an appropriate RVD-Demonstration-Flight. Up to the present, no such operation mission was specifically given, so recourse had to be taken to a mission model.

Figure 2.1 depicts the nominal profile of a LEO RVD flight, as part of a possible operational mission model. The trajectory shown is that of the chaser, with indicated numbers refering to main operations of Table 2.2. From No. 1.2.1 onwards, the RVD flight is shown in a target-centered, earth pointed reference frame, but not to scale. The typical RVD phases are as follows :

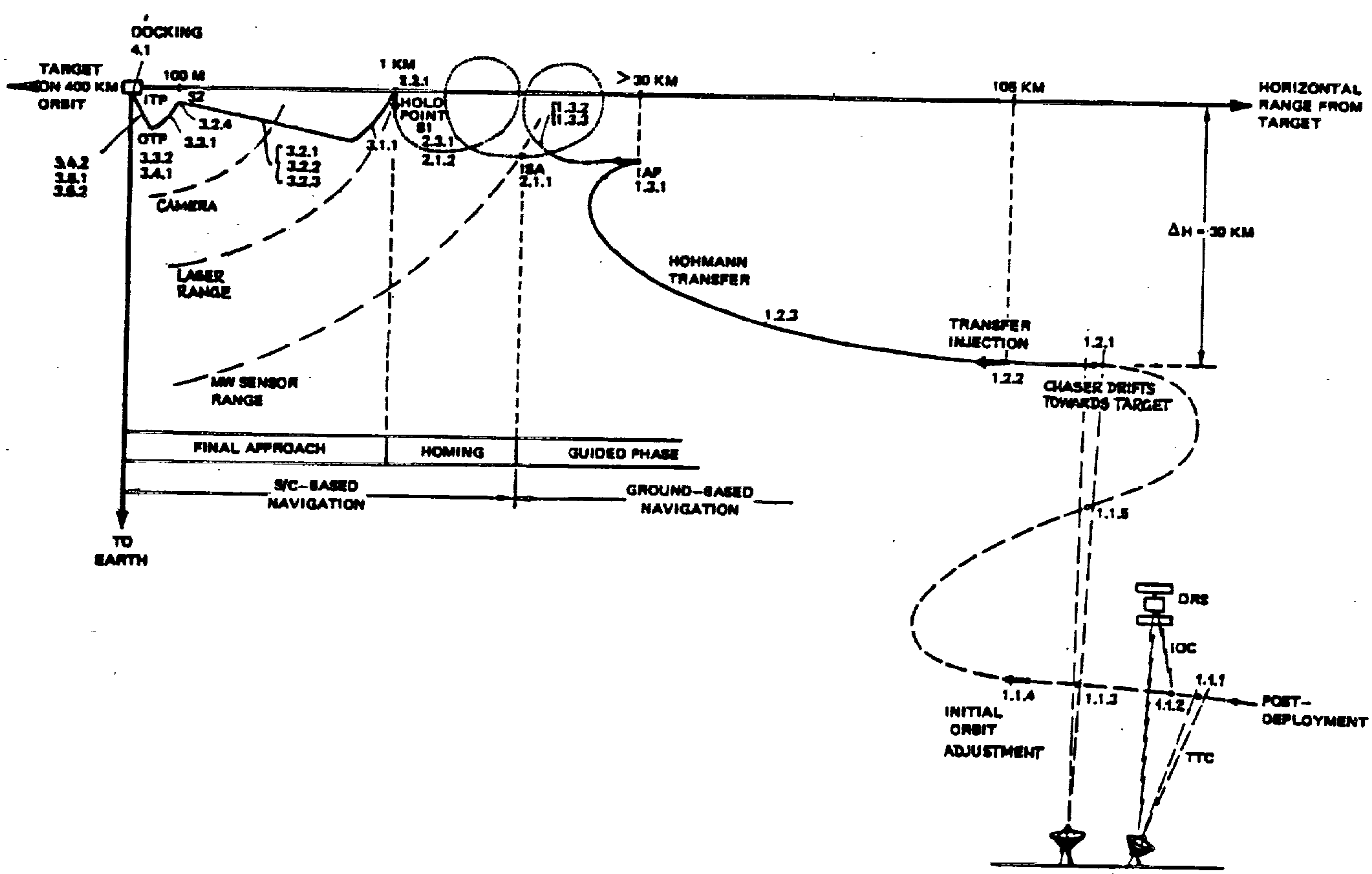

Figure 2.1 : The nominal profile of a possible LEO Rendezvous Mission Model

Ground Guided Phase :

This mission model assumes an operational target spacecraft, earth pointed, on a 400 km circular LEO. The chaser is deployed on a somewhat lower, nearly coplanar orbit. After ground contact establishment and chaser health check, its orbit is adjusted, under-ground command, to rendez-vous-compatible initial conditions, enabling subsequent inter-s/c-acquisition (ISA). From there on, the chaser takes over its own navigation and guidance.

Homing Phase :

The subsequent flight is the autonomous RVD proper. Based on relative position measurements of its own navigational sensors, the chaser guides itself along the "walking ellipse" and manoeuvres into a first "hold point" in the vicinity of the Target. This hold point serves for check-out purposes, verification of medium or short range sensors and/or for waiting for more convenient orbital constellation for the subsequent final approach.

Final Approach Phase :

At the proper point in time, the chaser departs from the hold point and following selected guidance schemes, such as proportional navigation and circumflight of the target, the chaser further approaches the latter and acquires its docking axis, respectively, all under its own GN & C.

Once on the Outer Target Point (OTP), on the Target docking axis, the chaser continues its terminal closure, along the docking axis, towards the Inner Target Point (ITP), just in front of the Target docking port, performing proximity operations in preparation of the subsequent Docking itself (Ref. 2).

Rendezvous Phase	No. in Fig. 2.1	Main Operation
Ground-Guided Phase	1.1.1 1.1.2 1.1.3 1.1.4 1.1.5	Establishment of TTC link Check on link to Data Relay Satellite Orbit determination from ground Orbit adjustment Tracking and thruster calibration
	1.2.1 1.2.2 1.2.3	Orbit transfer preparation Transfer thrusting Preparing Initial Aim Point (IAP) thrusting
	1.3.1 1.3.2 1.3.3	Acquisition of "Walking Eclipse" Coasting towards target Preparing for inter-s/c-acquisition (ISA)
Homing Phase	2.1.1 2.1.2	Target acquisition by chaser sensors Autonomous GN & C initiation, check by ground
	2.2.1	Hold point S_1 acquisition
	2.3.1 2.3.2	Position maintenance Preparing for Final Approach
Final Approach Phase	3.1.1	Acceleration towards target
	3.2.1 3.2.2 3.2.3 3.2.4	Stabilization of line-of-sight (LOS) Range Rate vs. Range Control Fine braking Acquisition of Zero-Range-Rate Point, S_2
	3.3.1 3.3.2	Sustaining constant range Acquisition of Outer Target Point (OTP)
	3.4.1 3.4.2	Preparing for Terminal Closure Close in and Proximity Operations
	3.5.1 3.5.2	Final braking to inner Target Point (ITP) Preparing for docking (4.1)

Figure 2.2 : The Mission Model Rendezvous Phases and their Main Operations

Associated with this RVD mission model are particular on-board functions and hardware. Fig. 2.3 gives a possible basic block diagram of the chaser information management system, incorporating an RVD-dedicated processor and its affiliated instrumentation as a subsystem on its own. The instruments are listed in Figure 2.4, for both s/c involved, because

- navigational sensors and their target reflectors
- both s/c docking adapters
- both s/c utility liner couplings

must be functionally and physically matched to each other.

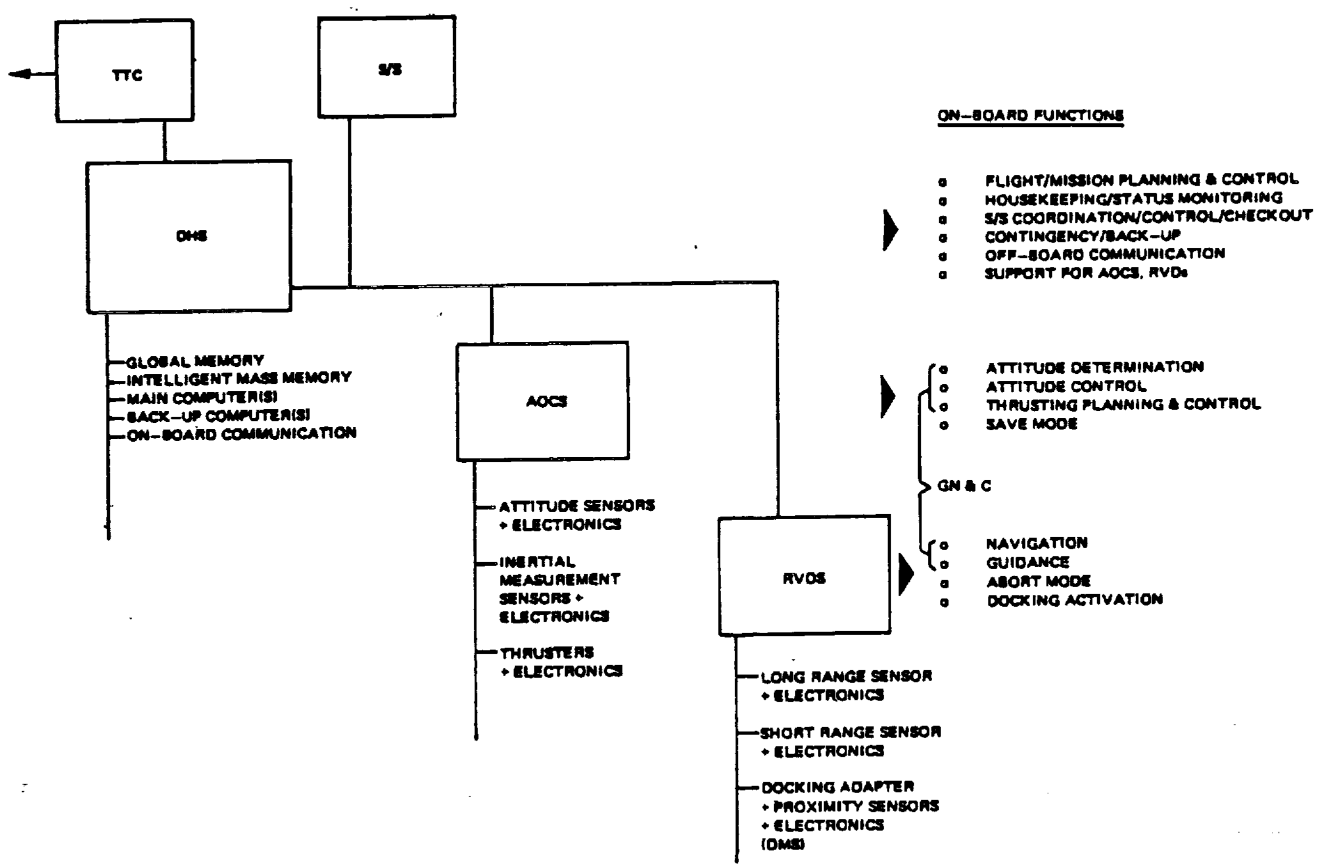

Figure 2.3 : Basic Diagram of the Chaser Information Management System

	TARGET S/C	CHASER S/C
LONG RANGE NAVIGATION MEDIUM RANGE NAVIGATION CLOSE RANGE NAVIGATION ATTITUDE MEASUREMENT	MICROWAVE (S-BAND) TRANSPONDER LASER RETRO-REFLECTORS OPTICAL TARGET MARKINGS SUN SENSORS, COARSE / FINE IR EARTH SENSORS GYRO PACKAGE	MICROWAVE (S-BAND RADAR LASER TRANSMITTER / RECEIVER CCD VIDEO CAMERA SUN SENSORS, COARSE / FINE IR EARTH SENSORS GYRO PACKAGE
ATTITUDE CONTROL	1 N NITROGEN COLDGAS JETS	10 N BI-PROPELLANT THRUSTERS 1 N NITROGEN COLD GAS JETS
VELOCITY MEASUREMENT		ACCELEROMETERS
VELOCITY CONTROL	20 N HYDRAZINE THRUSTERS	400 N BI-PROPELLANT THRUSTERS 50-80 N BI-PROPELLANT THRUSTERS 1 N NITROGEN COLD GAS JETS
DOCKING	PASSIVE DOCKING PORTS (HANDLES)	ACTIVE DOCKING MECHANISM (LATCHES)
SIGNAL LINES COUPLING	CONNECTOR SOCKETS	CONNECTOR PLUGS AND ACTUATOR
POWER LINES COUPLING	CONNECTOR SOCKETS	CONNECTOR PLUGS AND ACTUATOR
FLUID TRANSFER COUPLING	CONNECTOR SOCKET	CONNECTOR PLUG AND ACTUATOR

Figure 2.4 : Main RVD-related Equipment

Finally, the overview in Figure 2.5 correlates navigational sensors to the RVD phases and ranges and indicates necessary measurement accuracies. As can be seen, three types of "viewing" sensors have been incorporated, namely :

- a microwave pulse radar for far-range working e.g. in s-band
- a scanning laser radar for medium ranges
- a CCD camera sensor for short ranges

For more details about these sensors see Reference 3.

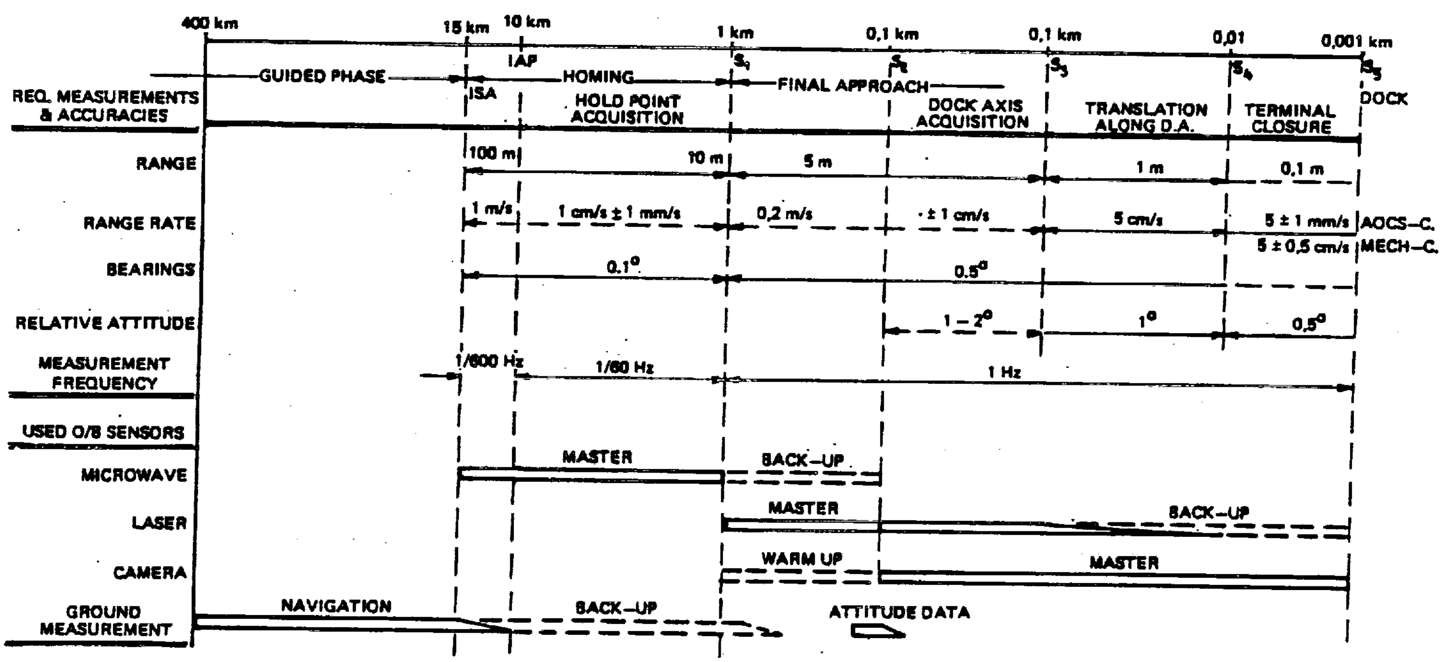

Figure 2.5 : Overview of Sensor Application Ranges

3. IN–FLIGHT TESTING OBJECTIVES

Individual RVD techniques, hardware and procedures, that do not lend themselves either easily or sufficiently convincing to qualification on ground, need validation in flight. Candidates for flight validation must comply with the following criteria :

- o have direct relationship with the RVD process or elements involved, such as sensor functions, GN & C modes, docking/de-docking. On the other hand, for example, communication via a data relay satellite, although of help in LEO-RVD-mission monitoring, is no direct part of the RVD process.
- o necessitate space environment and operations, such as laser sensor performance in vacuum on deep space background.
- o represent delicate operations under interaction of several subsystems and s/c dynamics and/or taking account of mission constraints, as for example, the interplay of the chaser attitude control system (AOCS) and Data Handling System (DHS) during AOCS reconfiguration following docking.

Applying these criteria within the analysis of the aforementioned mission model, performed in regard to in-flight manoeuvres and operations, vehicle hardware, software and on-board functions as well as ground segment availability and involvement, resulted in the following general categories of in-flight testing objectives.

- o Establishing space environmental effects in order to provide basic design data for major components. Figure 3.1 compiles some of these, whilst also indicating possible methods of flights.

COMPONENT	EFFECT	METHOD
CCD–ARRAY LASER DIODE	PERFORMANCE DEGRADATION DUE TO RADIATION ENVIRONMENT	TEST PACKAGE ON RETRIEVABLE MISSION, E.G. – EURECA MISSION – LDEF MISSION 6 – 12 MONTHS EXPOSURE TIME
LASER SENSOR CAMERA SENSOR	FUNCTIONAL PERFORMANCE UNDER VARIABLE ILLUMINATION CONDITIONS AND DIFFERENT RANGES, ANGLES. SPECULAR REFLECTION	TEST PACKAGES ON RETRIEVABLE MISSION, E.G. SENSOR ON ORBITER, REFLECTORS AND LED TARGET MARKING ON EURECA OR SMALL TARGET BODY. OR TEST PACKAGES ON EXPENDABLE MISSION, E.G. SENSOR ON APEX TECHN. SAT. TARGET MARKING ON AR–4, 3. STAGE
OPTICAL REFLECTORS	CONTAMINATION DUE TO PLUME IMPINGEMENT	TEST PACKAGE ON RETRIEVABLE MISSION, E.G. EURECA MISSION REFLECTORS MOUNTED ON OUTER PANELS. SHARED NASA/ESA INTEREST
RADAR SENSOR	FUNCTIONAL PERFORMANCE	TEST PACKAGE ON ORBITER. WITH TRANSPONDER ON EURECA OR APPROPRIATE TARGET BODY.
GUIDANCE CONTROLLER	FUNCTIONAL PERFORMANCE "SOFT ERROR"	TEST PACKAGE ON EURECA SIMULATING HOMING DURING THE GROUND CONTROLLED TRANSFER PHASES
DOCKING MECHANISM	AGING EFFECTS COLD WELDING	TEST PACKAGES ON LONG DURATION. RETRIEVABLE FREE–FLYER

Figure 3.1 : Establishing Space Environment Effects

- o Validation of mission phases performance, directly relevant to the automated RVD, namely : inter-s/c-acquisition, homing manoeuvring, hold point acquisition and maintenance, final approach inclusive terminal closure and proximity operations, docking and possibly un-docking and retreat.

- o Validation of performing the nominal mission strategies, included in the above phases, within given time constraints as determined e.g. by the availability of ground station links, launch and recovery system, demonstration vehicle constraints, representative lighting conditions etc.

- o Validation of the RVD spacecraft performances and capabilities in fulfilling system requirements under real environment conditions, including disturbances, such as propellant sloshing, atmospheric drag, plume impingement, electromagnetic interference from ground as well as when accounting for inaccuracies in orbit determination and errors in manoeuvre executions.

- o Validation of performance of RVD on-board operations of sensorics, GN & C, data handling etc, under control of on-board software or under ground control, including initiation of modes, their sequencing or transition switching.

- o Demonstration of the reliability and safety of the mission. This involves performance of selected back-up modes and contingency procedures, e.g. simulated contingency resulting in an abort of the final approach and manoeuvring back to a stand-by hold point.

- o Validation of post-docking activities, including AOCS reconfiguration, connection of utility lines.

- o Verification of the RVD analyses, ground-based simulations as well as ground preparation and mission support.

When translating the above objectives into individual qualification elements, the groups of Figure 3.2 result.

o OPERATIONAL MODES AND FUNCTIONAL PERFORMANCE UNDER DIVERSE ENVIRONMENTAL CONDITIONS AND RELATIVE POSITIONS OF :	⇨	– MW SENSOR – LASER SENSOR – CAMERA SENSOR – INERTIAL SENSORS – DOCKING MECHANISM – UTILITY LINES COUPLING – RENDEZVOUS CONTROLLER – DATA HANDLING SYSTEM
o METHODS AND SOFTWARE OF CHECK-OUT, SELF–TESTING :	⇨	– DITO –
o MODES AND PERFORMANCE OF INTERACTION BETWEEN :	⇨	– DHS, GN & C, AOCS
o SOFTWARE FUNCTIONS OF :	⇨	– TARGET SEARCH, LOCK–ON, TRACKING MODES – SENSOR SELECTION, NAVIGATION MODES – GUIDANCE SCHEMES FOR HOLD POINT CONTROL FOR PROPORTIONAL NAVIGATION FOR TARGET CIRCUMFLIGHT FOR TERMINAL CLOSURE FOR ABORT – ATTITUDE RECONFIGURATION – ON BOARD CONTINGENCY MANAGEMENT – OPERATING SYSTEM
o SPACECRAFT DYNAMIC BEHAVIOUR DURING :	⇨	– HOLD POINT CONTROL MANOEUVRE – PROPORTIONAL NAVIGATION PHASE – TARGET CIRCUMFLIGHT MANOEUVRE – TERMINAL CLOSURE AND ABORT – DOCKING/DE-DOCKING AND BACKING OFF – ATTITUDE RECONFIGURATION OF JOINT/SEPAR. S/C
o GROUND OPERATIONS PERFORMANCE AS TO :	⇨	– QUICK ORBIT DETERMINATION/PROPAGATION PREDICTION – MONITORING/CONTROLLING RENDEZVOUS STRATEGIES – COMPATIBILITY VERIFICATION BETWEEN GROUND–BASED AND ON–BOARD ESTABLISHED DATA – ASSESSMENT OF EXECUTION DATA VS. TOLERANCES – VERIFICATION OF CORRECT ON–BOARD S/W LOADING – VALIDITY CHECKING OF TELEMETRED DATA – QUICK TROUBLE–SHOOTING, CONTINGENCY MANAGEMENT – HANDLING SIMULTANEOUSLY TWO S/L – DISCRIMINATED TRACKING OF TWO CHOSEN S/C

Figure 3.2 : Qualification Elements

4. IN–FLIGHT TESTING APPROACH

Having identified what elements of the RVD process are to be flight qualified, the methods of performing the latter have been investigated and evaluated. As a result, a progressive step-by-step RVD qualification flight program (RQFP), culminating in and being round off by a single, final, dedicated RVD-Demonstration Mission, covering the whole process, emerged as safest and most cost-effective approach.

In the progressive program, individual elements, such as pieces of equipment, techniques and operational modes and procedures, are tested, verified and qualified in various steps, possibly one building upon the other and where repeated, building up confidence. Different approaches offer a variety of qualification methods. Figure 4.1 compiles some of these ; they include :

APPROACH	MEANS	PRIME VERIFICATION PURPOSE
VERIFICATION ELEMENT IS GIVEN A LIFT ON OTHER. NON–RVD–RELATED MISSION	TEST PACKAGE DEDICATED TO RVD VERIFICATION ELEMENT AS ADD–ON PASSENGER	o BASIC ENVIRONMENTAL IMPACT DATA o BASIC FUNCTIONAL PERFORMANCE
EXPLOITING OTHER S/C MISSION BY INCORPORATING RVD ELEMENT INTO FLIGHT PROFILE, SUCH AS: o FLIGHT ON TC INTO PRESET CONTROL BOX o TC RENDEZVOUS WITH IMAGINARY TARGET POINT IN SPACE	OTHER S/C SERVING AS PARTIAL TEST VEHICLE GROUND–COMMANDED "BLIND, NON–INTELLIGENT CHASER" (NO NAVIGATION SENSORS, NO GN&C AUTONOMY)	o BASIC MANOEUVERING ON TC o SINGLE S/C DYNAMICAL BEHAVIOR o PERFORMANCE OF GROUND SEGMENT o BACK–UP MODES
EXPLOITING OTHER S/C IN ORBIT AS NON–INVOLVED TARGET	GROUND–COMMANDED, DEDICATED TEST–VEHICLE, AS "NON–INTELLIGENT CHASER" (ON BOARD SENSORS, NO GN&C AUTONOMY)	o MANOEUVERING BASED ON GROUND–PROCESSED, ON BOARD MEASUREMENTS o PERFORMANCES OF SENSORS
	GROUND–MONITORED, DEDICATED TEST–VEHICLE, AS "INTELLIGENT CHASER" (FULL GN&C AUTONOMY, NO DMS)	o AUTONOMOUS GN&C o GROUND OVERRIDING PERFORMANCE o ABORT/CONTINGENCY PROCEDURES
TWO DEDICATED S/C MISSION WITH RVD–INVOLVED TARGET S/C	GROUND–MONITORED, DEDICATED TEST–VEHICLE AS "INTELLIGENT CHASER" (FULL GN&C AUTONOMY, DMS)	o COMPLETE RENDEZVOUS PROCESS o DOCKING/UNDOCKING PROCESSES o RECONFIGURATION OF S/C AOCS.

Figure 4.1 : Alternative Approaches to Flight Testing

- host missions to individual test packages or add-on experiments, mainly for basic data establishment or component functional tests.

- non-RVD-missions utilized for methods and procedures validation, such as manoeuvring modes, ground crew training.

- single test vehicle flights for s/c dynamical behaviour testing or contingency management routines.

- twin s/c mission with non-involved second s/c as target for rendezvous (no-docking) qualification only.

- twin s/c mission with fully involved second target s/c for complete rendezvous and docking demonstration.

Provided s/c availability and appropriate flight opportunities, careful allocation of qualification elements to the different flight approaches allows sequencing of steps and structuring of the RQFP to be characterized by the following :

- Basic design data and the environmental world are established first and in time to be incorporated into hardware and concept development

- Hardware components are development tested and space qualified individually, in conjunction with necessary ground testing and simulators involvement, prior to being incorporated into the system.

- Subsystems interactions and system control software are qualified progressively with increasing complexity.

- Ground-based capability of handling two RVD Spacecraft and of overriding chaser own decisions is built up and rendered routine prior to letting the chaser in space on its own.

- Contingency management and abort modes are qualified prior to attempting full RVD and endangering the s/c.

- Rendezvous modes are qualified in the sequence of decreasing ranges, thus of increasing risk to the target. Far range flight concept comes first, further approaches to the target follow the gain in confidence in the steerability and controlability of the chaser.

- Docking is attempted last, only after adequate confidence in rendezvous and retreat has been established.

- Ample time is offered for development of an optimally designed, dedicated RVD demonstration chaser.

An RVD qualification flight program thus defined

- o progressively builds up the level of confidence in the RVD concept and its elements
- o ends up with system elements largely space qualified
- o diminishes the need for on-board testing and telemetry facilities
- o renders the ultimate RVD-demonstration mission less complex, less risky and as such a final rehearsal prior to the operational application of the RVD concept.

Figure 4.2 shows parts of such an RQFP, albeit an elder concept dated mid 1984. It shows program links, utilization of non-RVD flight opportunities and ends up with Eureca 2 flight as the final RVD-Demonstration-Mission.
It should be noted that dates and flight opportunities have changed in the meantime, rendering the shown program obsolete.

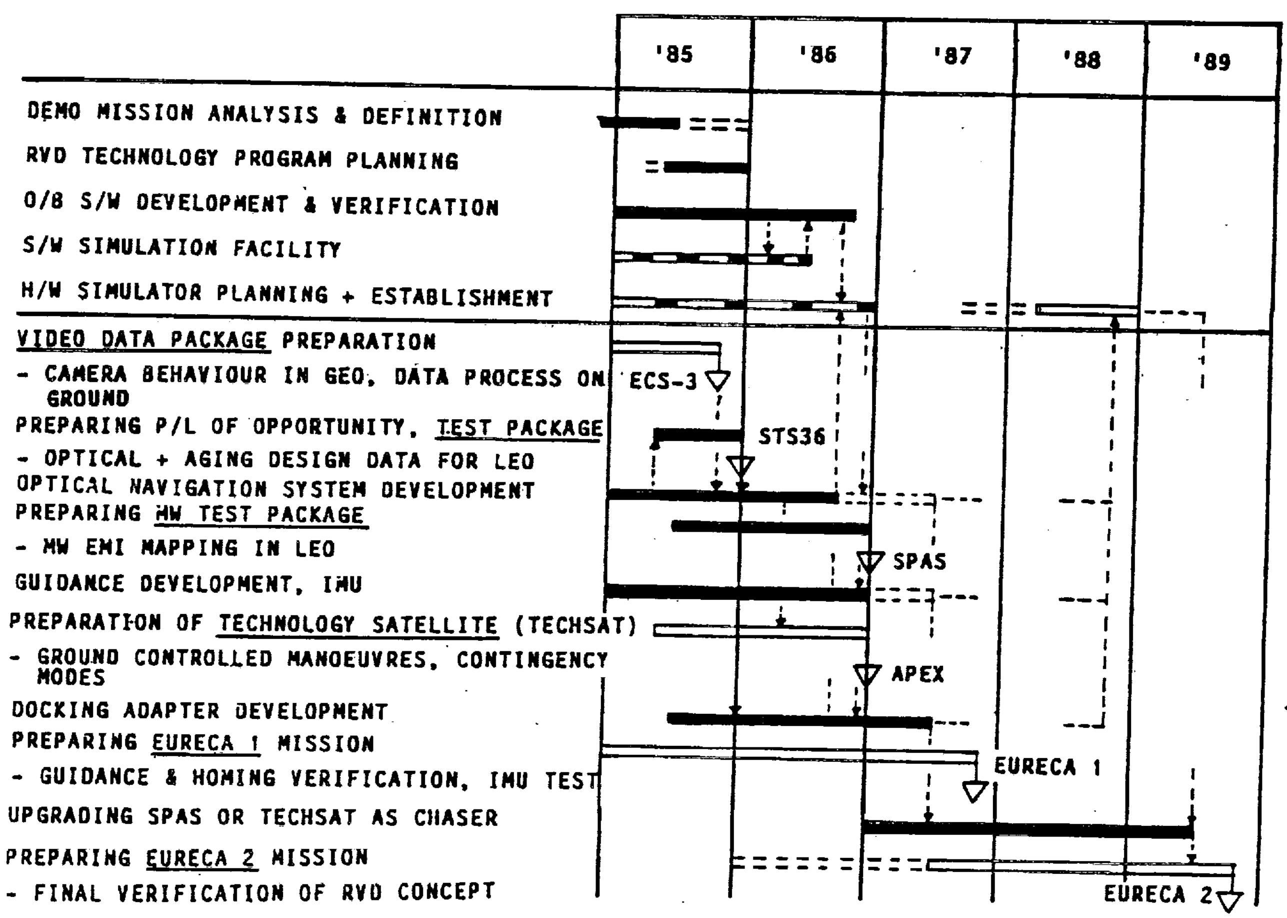

Figure 4.2 : Sample of an RVD Qualification Flight Program

The flight hardware that can be utilized for flight qualification purposes may be diverse, see Figure 4.3. To some degree it depends on the time horizon envisaged. In principle, for verification up to the end of the present decade, those termed "category '80" could be used, provided flight opportunities are identified and verification planning and implementation are timely initiated. For verifications up to, say, the middle of the 90-ies, those termed "Category '90" may be conceived in addition, again under the condition that their development will be decided upon.

As shown, most of the available flight hardware is Shuttle-borne. But expendable launch vehicles should not be excluded from consideration. They may offer attractive opportunities of flight, in particular for short-duration add-on test items, where recovery is not mandatory for retrieval of results.

VERIFICATION ITEM AS:		FLIGHT HARDWARE APPLICABLE FOR IN-FLIGHT VERIFICATION	
		CATEGORY '80: EXISTING OR IN DEVELOPMENT (FIRST OPPORTUNITY)	CATEGORY '90: POSSIBLE NEW DEVELOPMENT OR PROCUREMENT
SHUTTLE-BORNE	o SORTIER PAYLOAD INSIDE CARGO BAY (DAYS ON LEO)	o "GET-AWAY" (GAS)-CANISTER/NASA (1986) o "MAUS"-CANISTER (1986) o HITCHHIKER-G CARRIER/GSFC (1987) o HITCHHIKER-T CARRIER/TELEDYNE (1988) o SPACELAB PALLET/NASA (1986) o SPACELAB MODULE/NASA (1986)	o EUROPEAN SPACELAB PALLET/ESA
	o SORTIER PAYLOAD DEPLOYED OUT OF CARGO BAY (DAYS ON LEO)	o TIED TO REMOTE MANIPULATOR SYSTEM NO. 1/NASA (1985) o TIED TO TETHER SYSTEM/NASA	o TIED TO RMS NO. 2/NASA o TIED TO EXTENDED BOOM OR MAST MOUNTED ON PALLET-LIKE STRUCTURE
	o DEPLOYABLE PAYLOAD (DAYS, WEEKS, MONTHS ON VARYING LEO)	o EURECA FLIGHT UNIT 1/ESA (1987) o SPAS-1 FREE-FLYER/MBB/ERNO o RETRIEVABLE ORBITING BUS (ROBUS)/DOSY (1988) o VACOSAT BUS/MBB/ERNO (1988) o LONG DURATION EXPOSURE FACILITY (LDEF)/NASA (1986) o LEASECRAFT SPACE PLATFORM/FAIRCHILD o TIROS S/C BUS/RCS	o EURECA FLIGHT UNIT 2 OR FOLLOW-ON/ESA o SPAS FOLLOW-ON o ANY FREE-FLYER AS DEDICATED CHASER o ANY SUB-SATELLITE AS TARGET o COLUMBUS PRESSURIZED MODULE (PM)/ESA o COLUMBUS PAYLOAD CARRIER (PC)/ESA
ARIANE-BORNE	• SEPARATED PAYLOAD ON FLIGHT OPPORTUNITIES (HOURS ON GTO)	o SPOT S/C BUS/MATRA o TECHNOLOGY SATELLITE/MBB/ERNO (1987)	o ANY TECHNOLOGY TEST SATELLITE
	o ON LAUNCHER ELEMENTS (MINUTES ON SUB-ORBIT OR HOURS ON GTO)	o ON 3rd STAGE/ARIANESPACE o ON JETTISONED BOOSTERS/ARIANESPACE	

Figure 4.3 : In-Flight Verification Means

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5. RVD DEMONSTRATION S/C AND ORBIT

Considering now the final RVD-Demonstration Mission, the selection of the used s/c, the type of orbit and the associated launcher system are interrelated. Starting with the type of orbit, the following arguments speak in favour of LEO as the preferred one :

- o RVD-in-LEO is expected to be the first European case of application of an automated RVD concept.
- o More LEO Spacecraft seem to be available thus earlier flight opportunities are expected.
- o In terms of environmental effects and operational constraints, RVD-in-LEO is the more demanding case. Mastering automated RVD-in-LEO will easily enable its application elsewhere.
- o Retrieval of demonstration s/c enhances post-mission evaluation and enables mission repetition with same hardware and s/c. This implies Space Shuttle involvement, binding the mission to LEO.
- o Space Shuttle involvement offers additional support services during the demonstration in LEO, such as flight monitoring, data relay or intervention in emergency case.
- o Finally, Space Shuttle involvement enables sharing launch cost with other payloads, thus reducing demonstration mission recurring cost. Also, the Shuttle is the only current vehicle offering routine injection into LEO within the next decade, maximizing mission initiation success.

A comprehensive morphological analysis of Shuttle-borne spacecraft, that could serve as the RVD demonstration vehicles, has been done. Figure 5.1 shows a few of them :

SPAS	–	has been flown in a low-performance version, thus must be upgraded to fly as a chaser
Eureca	–	its development has recently been decided, its first mission is in advanced planning (Ref. 4)
Sun-orientated Vehicle	–	is in an advanced defintion phase, shall serve as a standard core bus for satellites. Although conceived for Ariane launch it shall be flyable within the Orbiter too.
TODOS	–	was originally conceived as a chaser for RVD-in-GEO missions. The concept was dropped in the meantime.
Eureca Subsats	–	were especially conceived as least-cost demonstration targets, with Eureca as chaser.

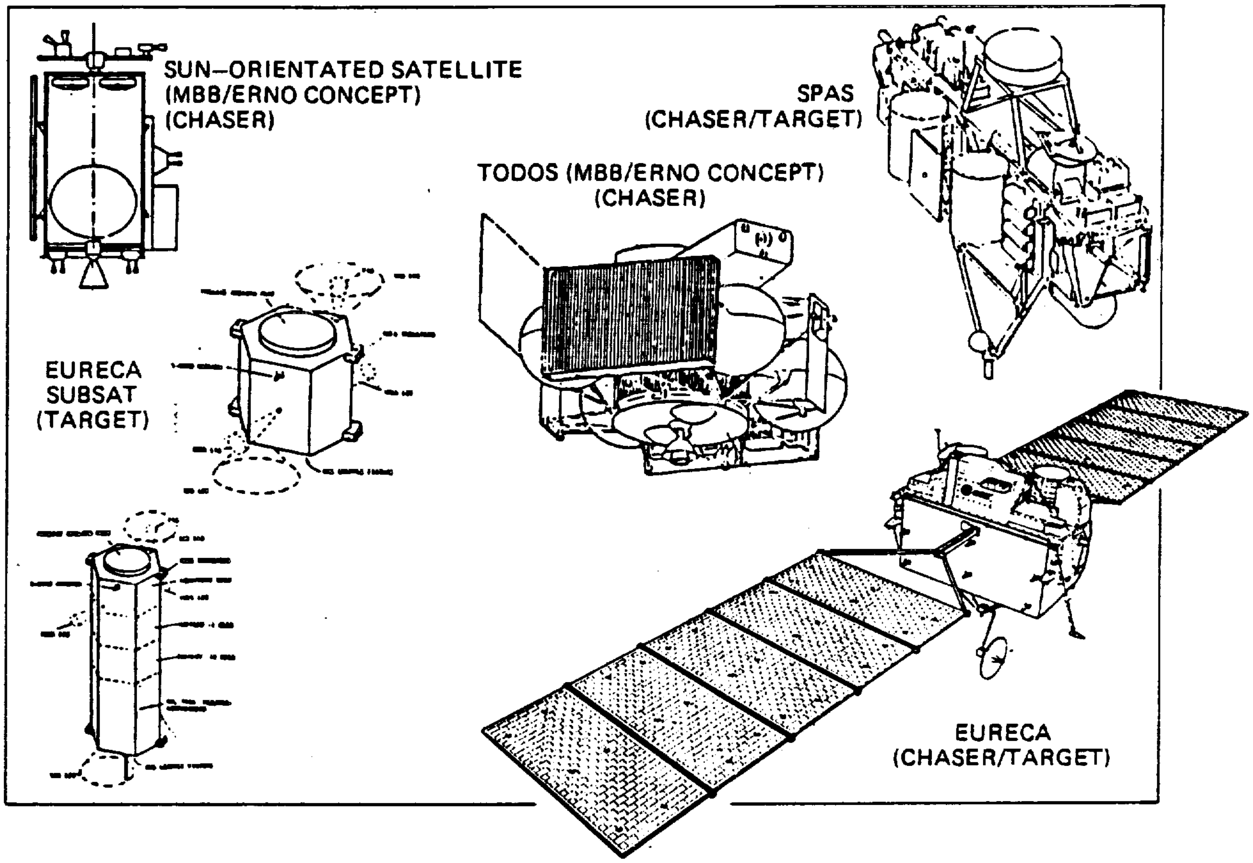

Figure 5.1 : Demonstration Spacecraft Alternatives

Now, to render the program viable in terms of cost and schedule, existing spacecraft should possibly be the first contenders, i.e. EURECA and SPAS, noting that for the time being only one flight unit apiece is foreseen.

To assign a role to any of them, mission requirements on chaser and target hardware must be established, then modifications necessary for assuming the role of a chaser or a target must be elaborated. That has been done for EURECA in several studies with the result that the choice of it as chaser induces important modifications in both the hardware and software areas. A major development interference of the two projects EURECA and RVD Demonstration would result, causing high risks for both.

As to cost, too, it is quite doubtful that the combination of

– a considerably modified EURECA as chaser, and
– a newly designed target, or adapted SPAS target

would be more interesting than the alternative of

– a little impacted EURECA as target, see Figure 5.2, and
– a considerably upgraded SPAS as chaser, or a newly designed chaser

The contrary seems more likely. For these reasons, the recommendation is to use EURECA as target s/c only and provide a dedicated chaser s/c either based on SPAS or on a completely new design. Figure 5.3 shows one such proposed chaser concept. In the present frame, possible chaser layout cannot be discussed further.

- VOLUME AND MASS
  - REQUIRED FOR RVD HARDWARE
  - LOST TO OTHER PAYLOAD

  IS MINIMUM
- ON–BOARD DATA HANDLING, TM/TC, SOFTWARE LOAD IS HARDLY AFFECTED BY RVD
- NO NEED TO COMPLICATE ON–BOARD SENSORIC
- LEAST DISTURBANCES TO EURECA MISSION PLANNING AND PREPARATION

LEAST COST CHARGES ON RVD DEMO

- RETAINS FULL ON–BOARD FUEL CAPACITY FOR OWN MISSION
- IF ABORT NECESSARY, EURECA IS TOO SLOW FOR PROXIMITY FLY BY OR HOLD
- IF SYSTEM FAILS DUE TO EMERGENCY SIMULATION DURING DEMONSTRATION, EURECA MISSION CAN CONTINUE AS PLANNED

ENSURE HIGHEST EURECA MISSION SUCCESS

- WITH DEMONSTRATION AT BEGINNING OF EURECA MISSION, RETRIEVAL OF RVD–INVOLVED HARDWARE IMMEDIATELY FOLLOWING DEMO IS POSSIBLE

IMPROVED LEARNING EFFECT

Figure 5.2 : Reasons for selecting EURECA as Target

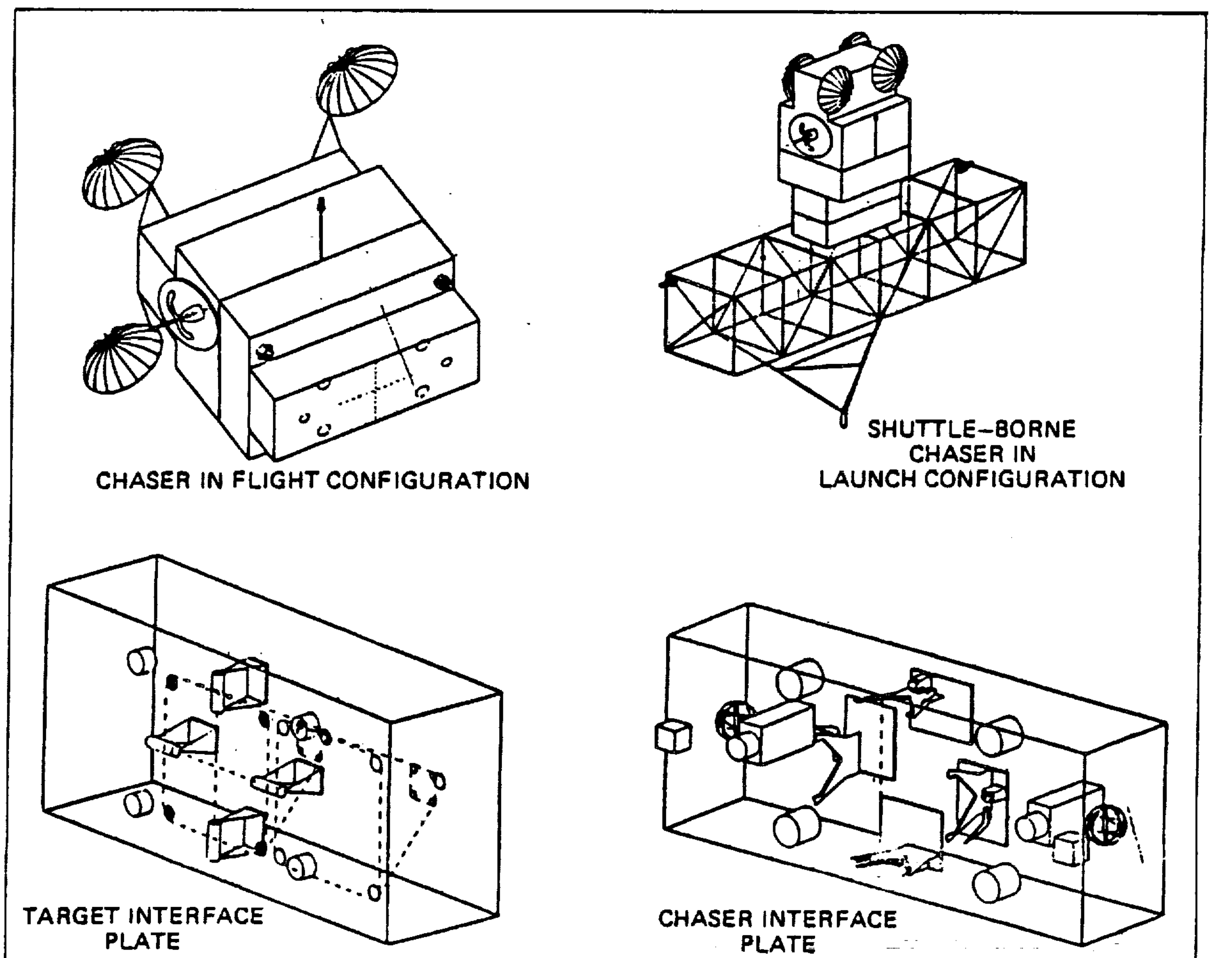

Figure 5.3 : A Chaser S/C Alternative Concept

Basically, there are various modes of coupling a chaser, such as a SPAS Version to EURECA, Figure 5.4. Investigations have shown the following merits for preferring the diagonal docking with the chaser length parallel to EURECA solar panels :

- o EURECA payload is impacted least
- o EURECA attitude sensors are not affected, Figure 5.5
- o The Orbiter RMS retains accessibility to EURECA grapple fixture, even in docked configuration
- o No deceiving effects by reflected sun light, from EURECA solar array, on the optical sensors of the approaching chaser
- o EURECA docking plane is inclined by 45° to the sun, so that there is no need to repeat docking demonstration once with the sun in the back of the chaser and once normal thereto.

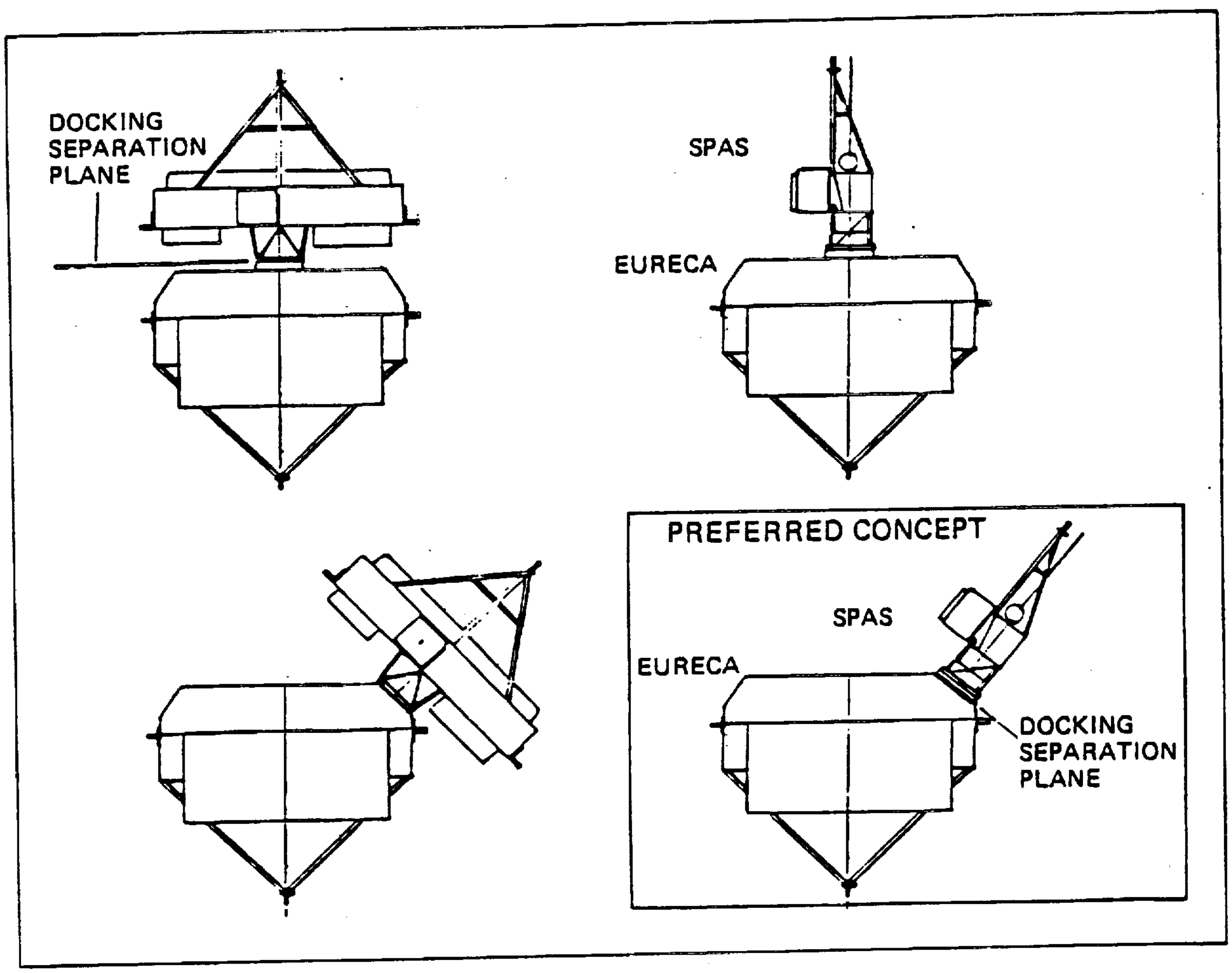

Figure 5.4 : Alternative Docking Position

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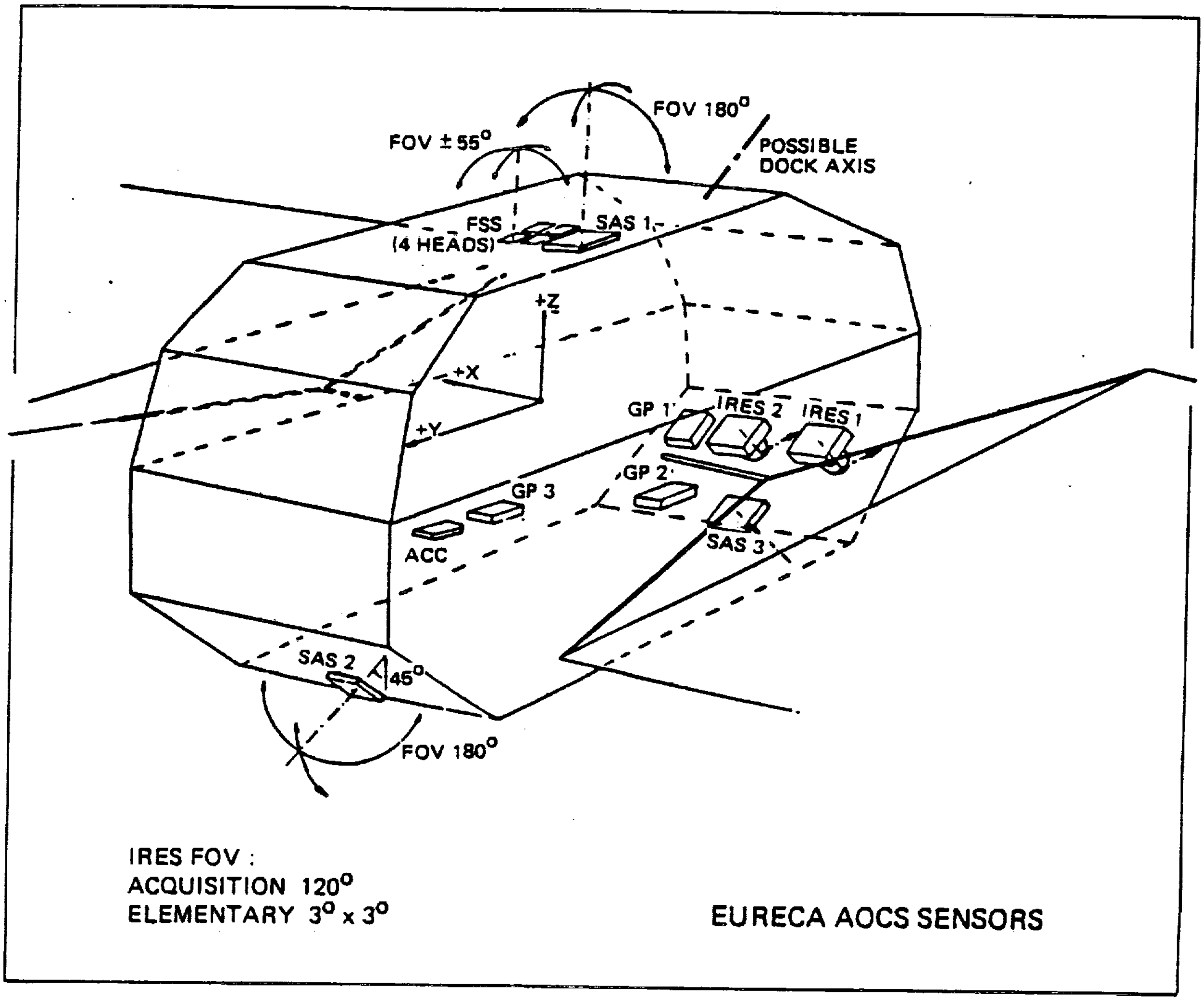


Figure 5.5 : EURECA Equipment

The expected drawbacks of

- o unsymmetry in docked configuration, is expected to be handled by reconfigured AOCS
- o docking impulse not going through EURECA e.g. is not considered critical, as resulting torque is very low.

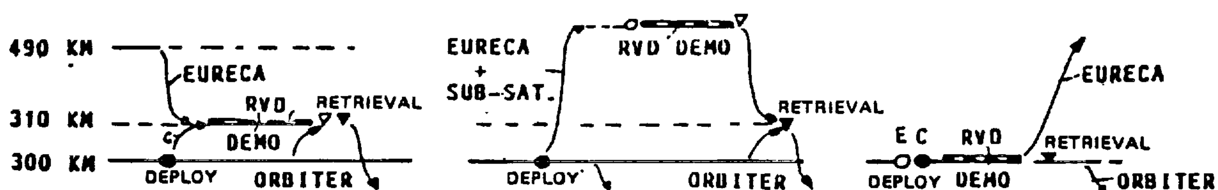
6. RVD DEMONSTRATION PROFILE

Different profiles of a Shuttle-borne, EURECA-utilized, RVD-Demonstration mission can be conceived. Figure 6.1 depicts and compares 3 alternative ones :

- after EURECA operational mission, prior to its retrieval
- during EURECA operational mission
- prior to EURECA operational mission, just after deployment

The last one is currently the preferred one. Its features are :

- o RVD Demonstration takes place on the Orbiter Standard orbit of 300 km altitude
- o RVD Demonstration takes place immediately following the deployment of EURECA (as Target) and the chaser (e.g. SPAS) from the Orbiter Cargo Bay, prior to EURECA departure onto its ascending transfer to its operational cell.



CRITERIA			
TARGET S/C	EURECA WITH MINIMUM AMOUNT OF MODIFICATIONS	SUB-SATELLITE IS A NEW DEVELOPMENT	EURECA WITH MINIMUM AMOUNT OF MODIFICATIONS
CHASER S/C	NEW DEVELOPMENT WHICH COULD BE USED FOR OTHER MISSIONS	EURECA WITH MAJOR MODIFICATIONS CAUSING IMPACT ON NOMINAL MISSION AND PAYLOADS	UPGRADED SPAS
DEMONSTRATION OPERATIONS	DEMONSTRATION PERFORMED AT THE END OF THE EURECA MISSION CHASER SYSTEMS ARE AT THEIR BEGINNING OF LIFE DEMONSTRATION PERFORMED OUTSIDE VISIBILITY RANGE OF ORBITER EURECA RVD HARDWARE 6 MONTHS IN SPACE ENVIRONMENT DEMONSTRATION PERFORMED AT END OF ORBITER MISSION	DEMONSTRATION PERFORMED AT THE END OF THE EURECA MISSION BOTH SPACECRAFT SYSTEMS 6 MONTHS IN SPACE ENVIRONMENT ORBITER NOT PRESENT DEMONSTRATION PERFORMED PRIOR TO CHASER END OF LIFE DEMONSTRATION PERFORMED BETWEEN ORBITER MISSIONS	DEMONSTRATION PERFORMED PRIOR TO NOMINAL EURECA MISSION CHASER SYSTEM ARE AT THEIR BEGINNING OF LIFE DEMONSTRATION PERFORMED IN VISUAL RANGE OF ORBITER ALL RVD SYSTEMS ARE AT BEGINNING OF LIFE DEMONSTRATION PERFORMED AT BEGINNING OF ORBITER MISSIONS
IMPLICATIONS	DEMONSTRATION LOST IF EURECA RVD SYSTEMS FAILED RISK OF LOOSING TRACK OF BOTH SPACECRAFT FOLLOWING A DEMONSTRATION ACCIDENT ORBITER TO PHASE ADJUST AND ASCEND FOR RETRIEVAL OF BOTH SPACECRAFT	RVD READINESS OF BOTH SPACECRAFT DOUBTFUL RISK OF LOOSING TRACK OF BOTH SPACECRAFT FOLLOWING A DEMONSTRATION ACCIDENT ORBITER TO PHASE ADJUST AND ASCEND FOR RETRIEVAL OF COMBINED SPACECRAFT	RVD READINESS OF BOTH SPACECRAFT ENSURED VISIBLE TRACKING FROM ORBITER FOLLOWING A DEMONSTRATION ACCIDENT CHASER ONLY RETRIEVED ON ORBITER MISSION ORBIT

Figure 6.1 : Alternative Demonstration Profiles

- o All RVD tests can be performed within sight of the accompaning Orbiter.
- o All RVD test objectives are covered, however in a sequence not fully coherent with the nominal run down of the mission model it intends to represent, i.e.
  - – near range objectives come first, including demonstration of emergency abort prior to docking
  - – docking with subsequent de-docking and possibly repetition of both processes being performed in midpoint
  - – long range objectives come last, when EURECA drifts away to embark on its nominal mission
- o The chaser can be recovered by the Orbiter immediately following achievement of RVD demonstration objectives

Figure 6.2 lists the associated run down. RVD demonstration starts already at event 7, when the target reaches a distance of about 2 km from the Orbiter and stays on a waiting ellipse while the chaser comes drifting towards it. At that point

- o ESANET takes over mission responsibility
- o Inter-s/c-acquisition by the laser and microwave sensor systems takes place.

EVT No.		Time h:min	Δv m/s	
1	EU	0:19	-	EU deployment from orbiter cargo bay
2	EU	1:19	-	EU checkout begin
3	SP	3:19	-	Begin SPAS checkout via umbilical
4	EU	3:35	0.123	Initiate EURECA drift manoeuvre
5	SP	4:51	-	SPAS deployment from cargo bay
6	EU	5:06	0.123	Stop drift manoeuvre at 2 km
7	EU	5:19	0.289	Initiate synchronous holdpoint between 2 - 3 km
8	SP	5:29	0.025	Initiate SPAS drift manoeuvre
9	SP	5:49	-	Establish inter S/C ranging by microwave
10	SP	7:16	-	Establish inter S/C laser ranging
				- test microwave & laser ranging -
11	EU	11:21	0.289	Stop EU oscillation at 2.0 km from orbiter
12	SP	11:31	0.025	Stop SPAS drift at 1.5 km from orbiter (4 orbits)
		13.50	e.o.e.	
13		14:02	0.180	Initiation of $\overline{R}$ manoeuvre - 500 to 125 m
		14:45	b.o.e.	
14		14:47	0.144	Initiation of a synchronous holdpoint 125 to - 125 m approximation to 63 m
		15:20	e.o.e.	
15		15:33	0.300	Initiation of rectinliear approach
16		15:41	0.400	Initiate LOS rotation at R = 50 m } $\dot{R} = 0$
17		15:44	0.200	Stop LOS rotation at R = 50 m
18		15:50	0.500	Initiate emergency program at R = 50 m
		16:15	b.o.e.	
		16:51	e.o.e	
19		16:55	0.300	Continue approach
20		17:02	0.100	Arrive at last meter, proceed to docking
21		17:04	-	Capture, docking, checkout
		17:46	b.o.e	
		18:21	e.o.e	
22		18:25	0.050	Initiate rotation by 45°/docking axis perpendicular to solar vector
23		18:47	0.050	Stop rotation, start checkout in second attitude
		19:16	b.o.e	
		19:52	e.o.e.	
24		19:55	0.100	Start de-docking & re-docking sequence
25		20:45		De- & re-docking sequence ended
		20:47	b.o.e.	
26		20:50	0.050	Initiate re-rotation 45° to previous attitude
27		21:15	0.050	Stop rotation
		21:22	e.o.e.	
28		21:25	0.100	De-dock & retreat to 20 m (rectilinear)
29		21:38	0.030	Initiate $\overline{R}$ manoeuvre transfer to 100 m
30		22:23	0.150	Initiate $\overline{R}$ manoeuvre transfer to -500 m
31		23:09	0.180	Stabilize holdpoint at -500 m
				- SPAS ready for retrieval -
32		23:28	(0.200)	Initiate EURECA retreat
				- Perform long range sensor testing -
33		28:30	-	EURECA reaches 10 km from SPAS -

EU - EURECA (Target)
SP - SPAS (Chaser)

b.o.e. - begin of eclipse
e.o.e. - end of eclipse

Figure 6.2 : RVD Demonstration Timeline

Figure 6.3 shows this initiation phase in an Orbiter-centered, earth-pointed reference system.

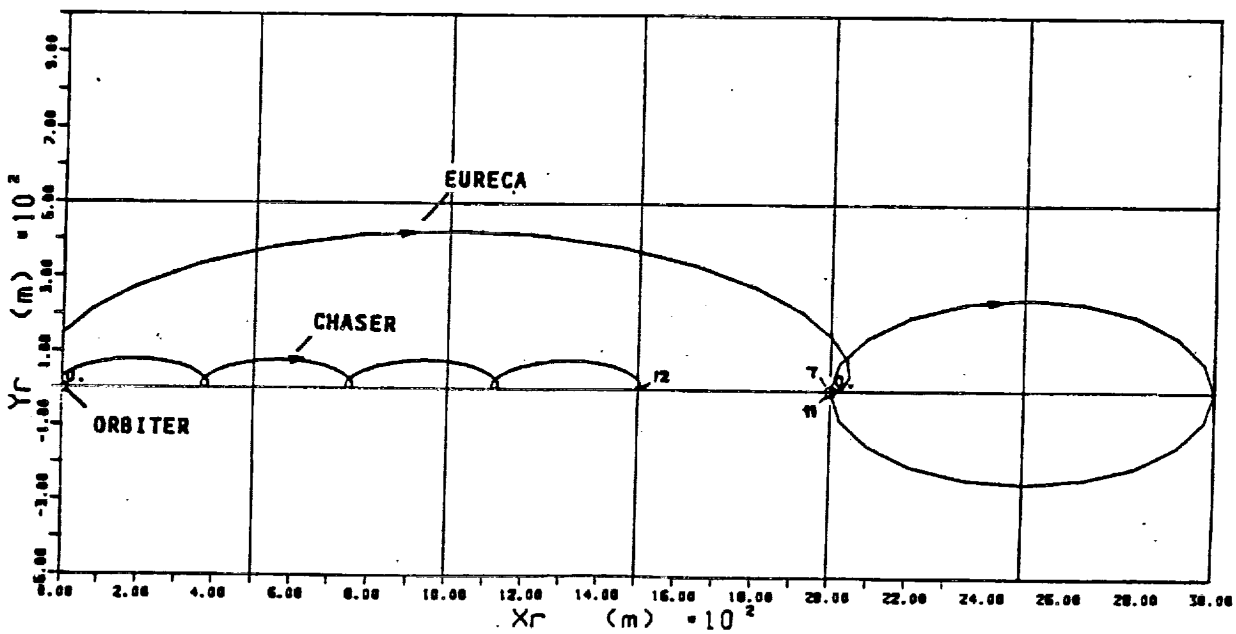
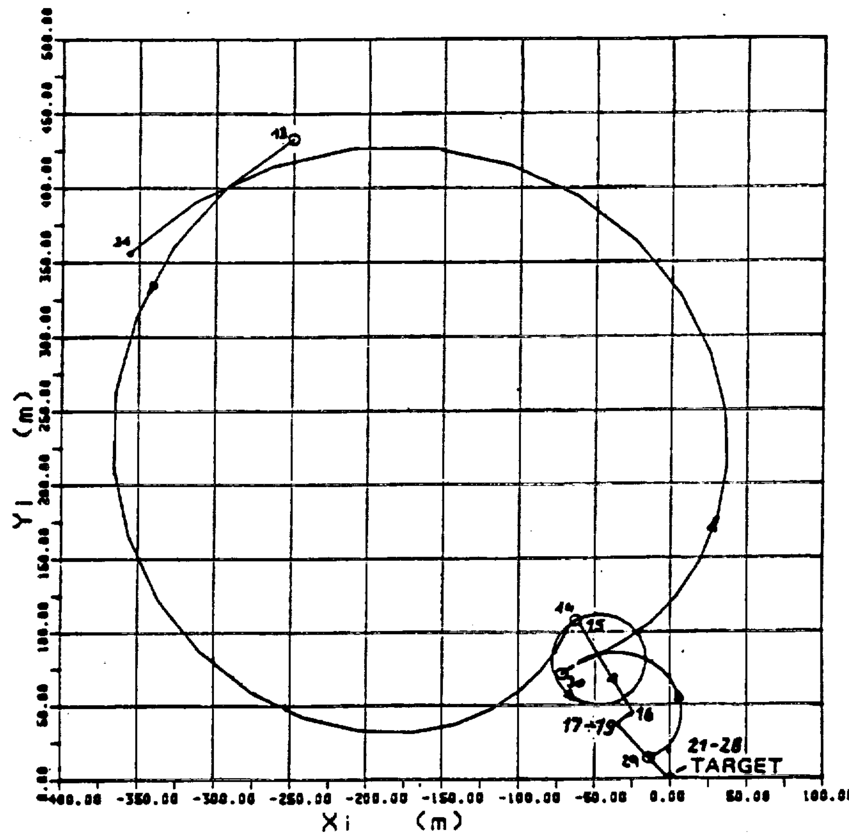


Figure 6.3 : S/C Drift Away after Deployment, seen from Earth Pointed Orbiter

Turning over to a target-centered reference system (index r denotes rotating, index i denotes inertial), Figure 6.4 shows the motion of the chaser around the target. Number refer to events in Figure 6.2. The following demonstrations take place :

Event	15 – 16	proportional navigation guidance scheme
	16 – 17	Los rotation at constant range of 50 m
	17 – 20	terminal closure plus emergency abort, retreat, retrial
	21	docking
	21 – 28	tests in docked configuration
	29 – 31	chaser retreat and acquisition of a hold point for retrieval

INERTIALLY ORIENTED FRAME



EARTH ORIENTED FRAME

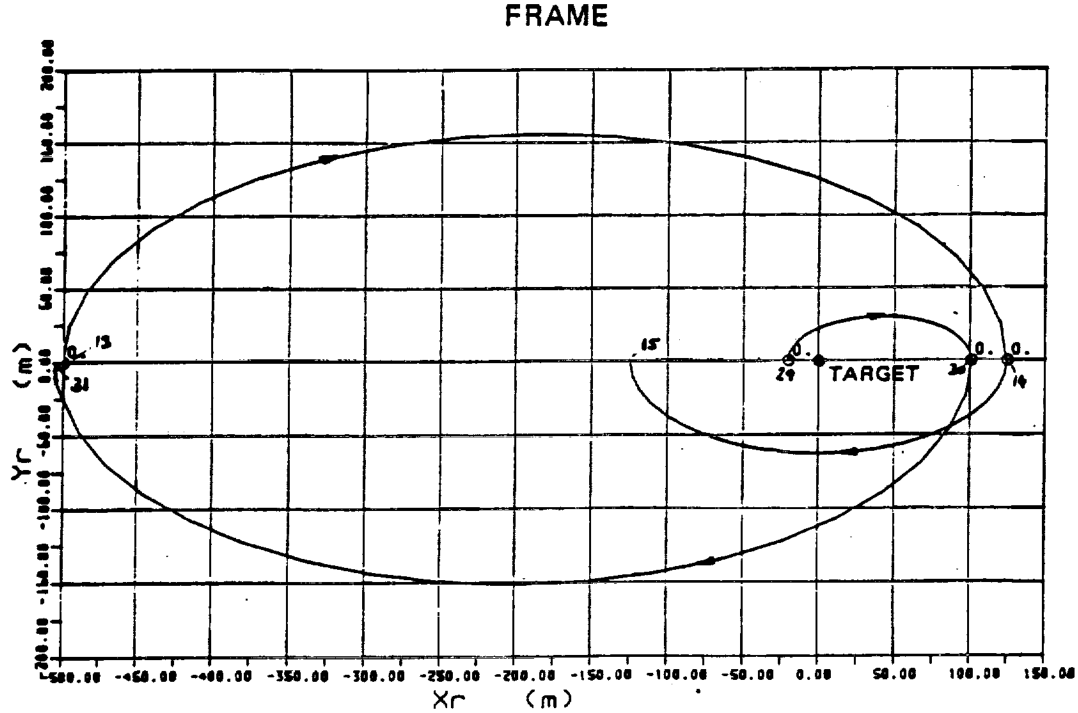


Figure 6.4 : Chaser Relative Motion in Target-fixed Frame

7. RVD DEMONSTRATION MONITORING

Finally, a short comment as to monitoring the Demonstration rundown. Successful achievement of demonstration objectives must be confirmed to mission control center, preferably in real time. In LEO, direct contact opportunities, s/c-ground are sparse and brief. Thus a data relay satellite (DRS), although not absolutely necessary for automated RVD – is of considerable help in sustaining links to ground. For EURECA missions a DRS link, via inter-orbit-communication (IOC) equipment aboard EURECA, is planned. Target monitoring is thus ensured.

For the chaser, alternative modes are possible, Figure 7.1. The preferred mode would be to modulate telemetry data on the microwave navigation link, from the chaser onto the target, and whence to ground.

The vicinity of the Orbiter during the rendezvous flight offers here a further nominal or back-up alternative.

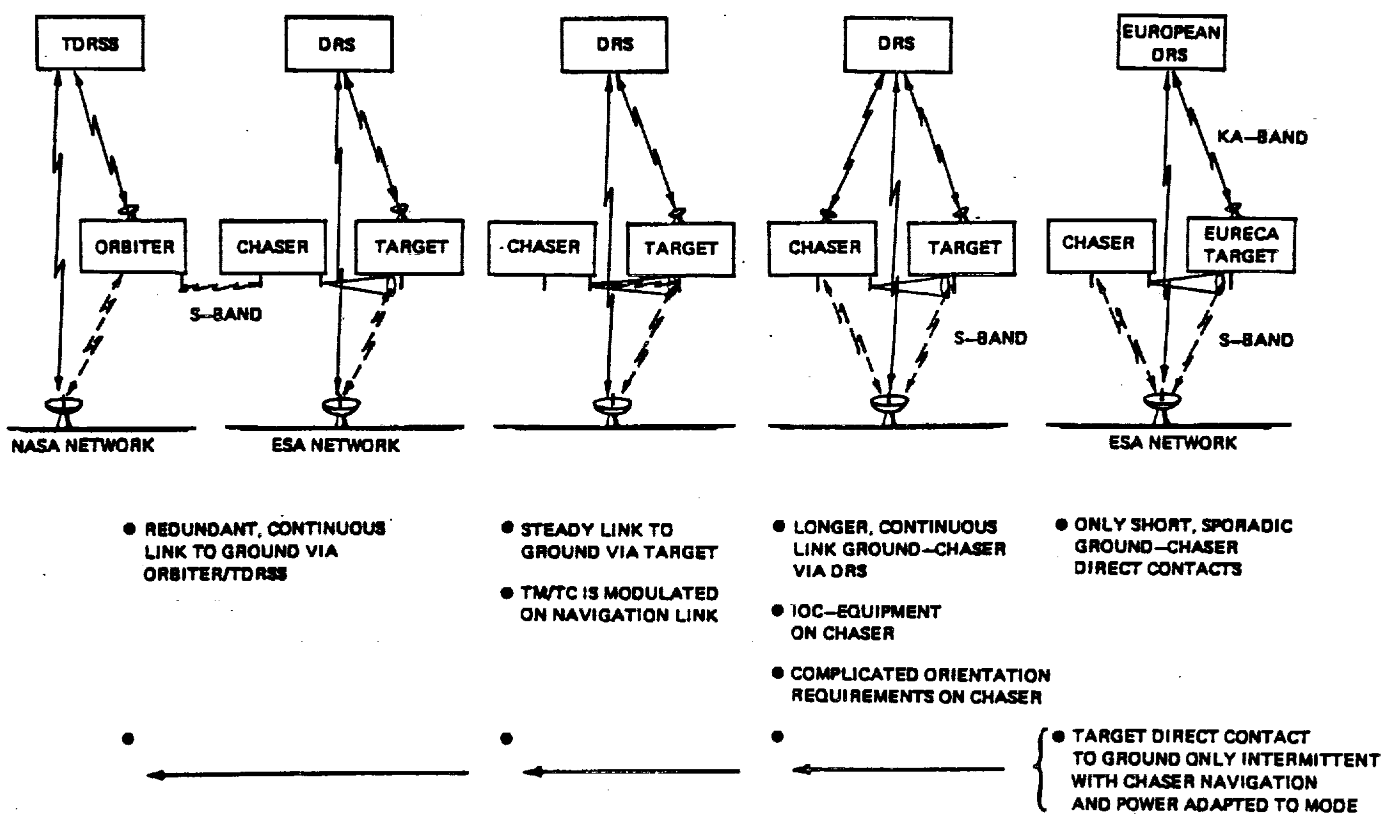


Figure 7.1 : RVD Demonstration Monitoring Alternatives

8. CONCLUSIONS

- The development of an automated Rendezvous and Docking capability calls for an in-flight qualification program that may culminate in a complete RVD-Demonstration Mission.

- Performing the RVD demonstration in LEO, as part of an EURECA flight, seems to be the more viable and economical way

- Choosing the EURECA platform as target s/c and developing a dedicated chaser s/c is currently the preferred solution

- The RVD-Demonstration Mission may come earliest at the beginning of the 90-ies

- The involvement of the shuttle in the RVD-Demonstration flight may be extended to advantage beyond the mere transportation functions, in particular for monitoring and as contingency data-relay back-up.

- A joint U.S. – European mission for demonstrating automated RVD is deemed an attractive venture.